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ADVANCED COMPOSITE STABILIZER FOR BOEING 737 AIRCRAFT

18 JULY 1978

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FOURTH QUARTERLY TECHNICAL PROGRESS REPORT
19 APRIL 1978 THROUGH 18 JULY 1978

PREPARED FOR:

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
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HAMPTON, VIRGINIA 23665

IN RESPONSE TO:

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DRL LINE ITEM NUMBER 018

BOEING COMMERCIAL AIRPLANE COMPANY

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Boeing Commercial
Airplane Company
Contract NAS1-15025

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FOURTH QUARTERLY TECHNICAL PROGRESS REPORT
19 April 1978 through 18 July 1978

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FOREWORD

This report was prepared by the Boeing Commercial Airplane Company, Renton, Washington, under Contract NAS1-15025. It is the fourth quarterly technical progress report covering work performed between 19 April 1978 and 18 July 1978. The program is sponsored by the National Aeronautics and Space Administration, Langley Research Center (NASA-LRC). Dr. H. A. Leybold is the Project Manager for NASA-LRC.

The following Boeing personnel are principal contributors to the program during the reporting period: G. Ohgi, Design; R. Johnson, Structural Analysis; M. Garvey, Manufacturing Specialist; D. Grant, Production Manager; L. D. Pritchett, Technical Operations Coordinator; and D. V. Chovil, Business Support Manager.

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SUMMARY

Activities related to development of an advanced composites stabilizer for the Boeing 737 commercial transport are reported.

Activities include discussion of criteria and objectives, design loads, the fatigue spectrum definition to be used for all spectrum fatigue testing, fatigue analysis, manufacturing producibility studies, the ancillary test program, quality assurance, and manufacturing development.

The fatigue load sequence was developed similarly to the European standard spectra, TWIST and FALSTAFF. Selection of a base mission for spectrum definition was accomplished by reviewing the original 737 analyses, and the 10 years of service history since the 737 was introduced into service. Design is proceeding with detailed design of graphite/epoxy components. Producibility studies on the rear spar/lug interface test section have been completed, and include the spar detail bonding, machining, and drilling, and attachment of the titanium lugs. The program is progressing as scheduled.

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SECTION 1.0

INTRODUCTION

The escalation of jet-fuel prices is causing a reassessment of technology concepts and trades used in designing and building commercial airplanes. The task is to incorporate fuel-saving concepts into commercial aircraft design.

The potential weight savings and fuel reduction resulting from the use of advanced composites in aircraft structure, especially primary structure, are significant. However, the lack of technical confidence and cost data has delayed their use in commercial aircraft.

Hardware programs conducted in a production environment are required to establish and demonstrate the safety, operating-life characteristics, and manufacturing cost of advanced composite primary structures.

Boeing's approach to the problem is to obtain reliable production, technical, and cost data bases by the integration of advanced composites technology development under NASA contracts, which, when combined with company effort, will accelerate the application of composites. This approach addresses these data bases, and develops realistic production costs in a commercial transport manufacturing environment. Program emphases are directed toward developing the information needed to obtain an early production commitment decision by management, and will be conducted in a production environment.

Preliminary developments, as covered in the first quarterly report, were devoted to conceiving, developing, and analyzing alternative design concepts, and the preparation of a technical plan to aid in selecting and evaluating material, identifying ancillary structural development test requirements, and defining full-scale ground-test and flight-test requirements necessary to obtain FAA certification.

The program was built on precontract design activities as well as contracted design activities that consider:

- Program management and plans development
- Establishing design criteria
- Conceptual and preliminary design
- Manufacturing process development
- Material evaluation and selection
- Verification test
- Detail design
- FAA approval plan definition

This report describes work accomplished during the fourth 3-month period of the contract. Design activities include discussion of the design loads, the fatigue spectrum and analysis approach, design details, producibility studies, and the ancillary test program. These activities are described under the headings: Design and Analysis, Development Test Plan and Status, and Operations Development. The overall schedule status is summarized in Figure 1-1.

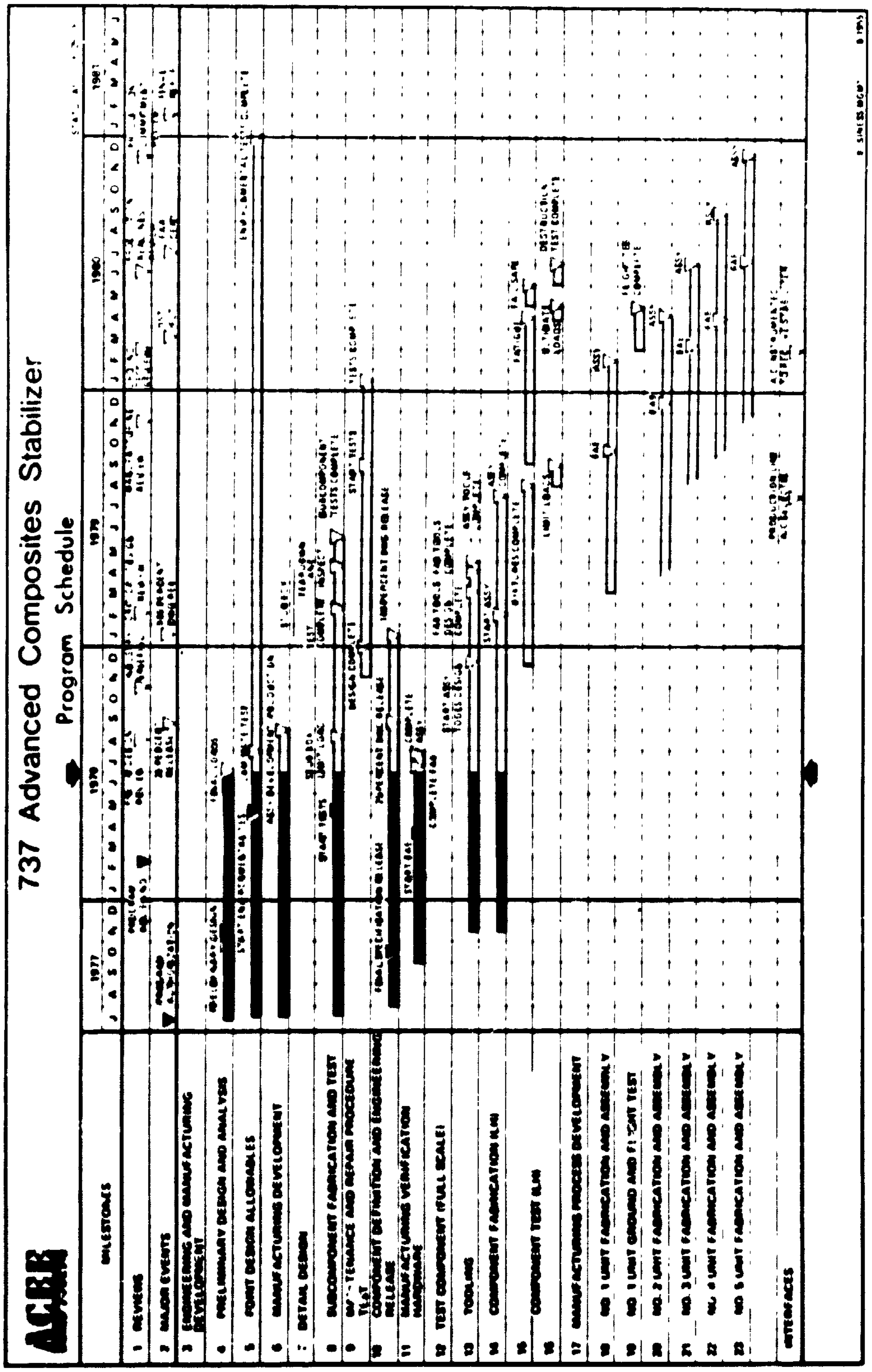

Figure 1-1. Program Master Schedule

SECTION 2.0

DESIGN AND ANALYSIS

2.1 DESIGN LOADS CRITERIA AND ANALYSIS

2.1.1 Criteria and Objectives

Preliminary design criteria and objectives are being established for the advanced composites horizontal stabilizer. A preliminary list of design criteria and objectives for this program, which are presently being finalized, is presented in Reference 1.

2.1.2 Design Loads

The horizontal stabilizer will be substantiated for the highest loaded model 737 airplane. Requirements of Federal Aviation Regulations (FAR) and Boeing design specifications will be met.

The three critical load cases that are presently being used for preliminary design are presented in Reference 1. Pressure loadings that are being used for local design and skin panel attachments are presented in Reference 2.

The fatigue spectrum definition to be used for all spectrum fatigue testing has been defined. The load sequence has been developed similarly to the European standard spectra TWIST and FALSTAFF (References 3 and 4), in which flights of varying severity are applied with more and larger load peaks in severe flights than in lesser flights.

Selection of a base mission for spectrum definition was accomplished by reviewing the original 737 fatigue analysis, and the 10 years of service history since the 737 was introduced into service. Existing fleet service utilization data were investigated. This information indicated that projected flights in 20 years will number approximately 50,000, for the median utilized aircraft. The average flight length of the median utilized aircraft is between 463 and 741 km (250 and 400 nmi). The 463-km (250-nmi) range was selected as the base mission, based on the fact that metallic fatigue damage per flight for the 737 spectrum has been shown to be constant between the 463- and 741-km (250- and 400-nmi) missions.

The 463-km (250-nmi) flight profile defined in the existing 737 fatigue analysis consists of 24 segments, each with 1-g gust and maneuver loads. The total flight profile has been reviewed. The test flight profile was reduced to six major flight phases, defined as taxi, takeoff, climb, cruise, descent, and landing. The taxi, takeoff, and landing phase alternating loads are of a relatively small magnitude, so these phases are represented by single excursions of the 1-g load, plus the secondary cycle excursion. Significant alternating load activity exists during climb, cruise, and descent phases, so these test phases will contain an appropriate number of alternating load peaks about the 1-g load levels. The resulting general load sequence is shown in Figure 2-1.

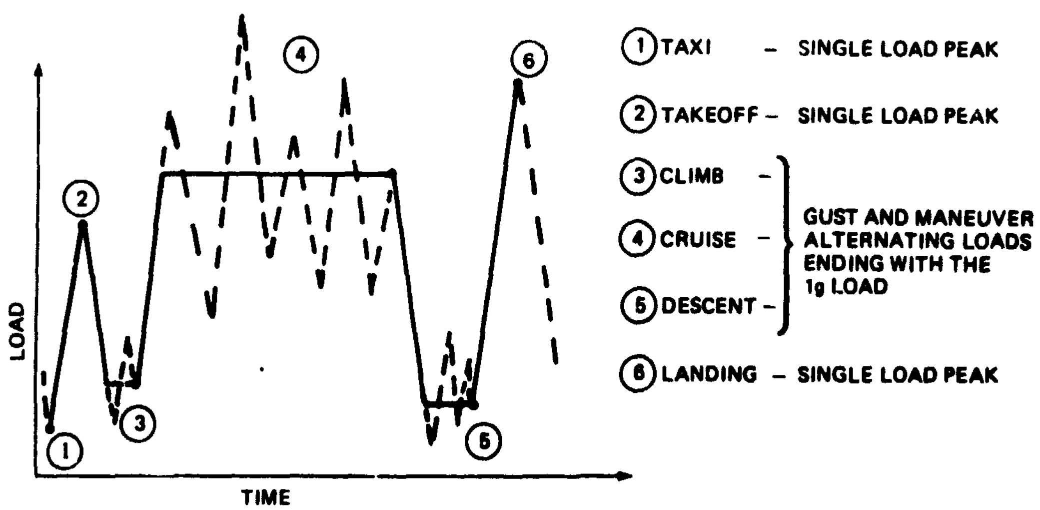

Figure 2-1. Test Spectrum General Loading Sequence

Prior to selecting the number and magnitude of alternating load peaks, the importance of small-cycle omission and large-cycle truncation was investigated. In previous graphite/epoxy fatigue testing, Schutz and Gerharz (Reference 5) used an omission level of 6% of ultimate as a baseline, and found that further omission resulted in life increase. Based on this, the omission levels were set at 6% of ultimate for maneuver, and 3% of ultimate for gust. This resulted in an average of 10 maneuver and seven gust load cycles per test flight, or an average of 20 load cycles per test flight, including the secondary GAG cycles.

Truncation load levels were examined in accordance with the standard spectrum TWIST (Reference 3), which truncates at the load level exceeded 10 times per lifetime. Schutz and Gerharz showed that truncation of the highest test spectrum loads to 90% had virtually no effect on the fatigue life of graphite/epoxy.

Based on this, truncation levels were conservatively set at the load exceeded five times per lifetime, which corresponds to approximately 90% of the load exceeded once in two lifetimes. Therefore, based on the previously defined 50,000 flights per lifetime, the test spectrum will be constructed from 10,000-flight blocks.

Eight gust and eight maneuver alternating load levels were defined, resulting in the stepped exceedance curves shown in Figures 2-2 and 2-3. Table 2-1 lists the resulting occurrences of gust and maneuver incremental loads to be applied in one 10,000-flight block.

Many of the alternating loads contained in the test spectrum occur less than once per flight, necessitating several test flight types with different severities and frequencies. Test flight severity levels were defined in a similar manner to those defined in TWIST, (Reference 3). Eight flight

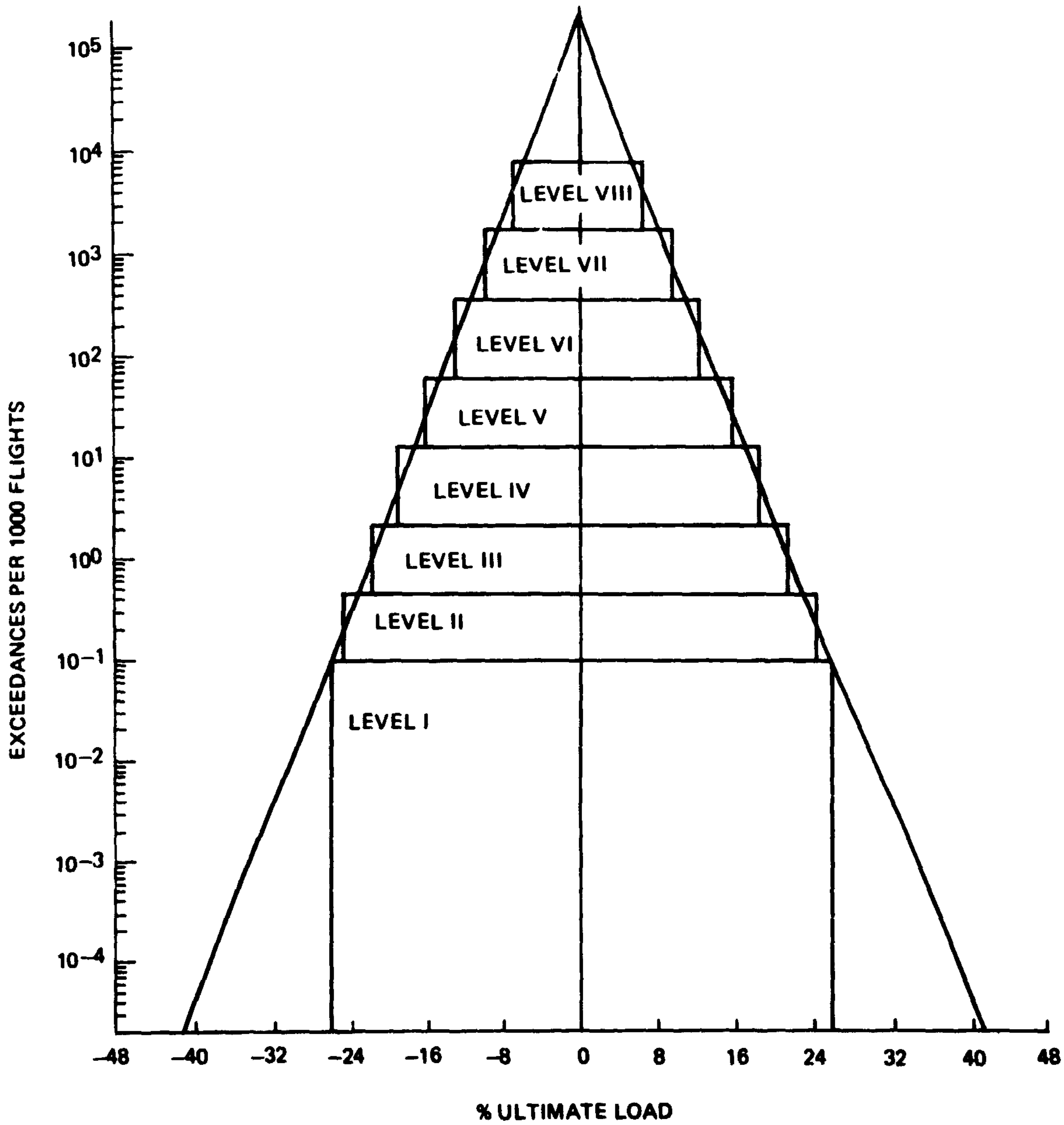

Figure 2-2. Maneuver Alternating Load Levels

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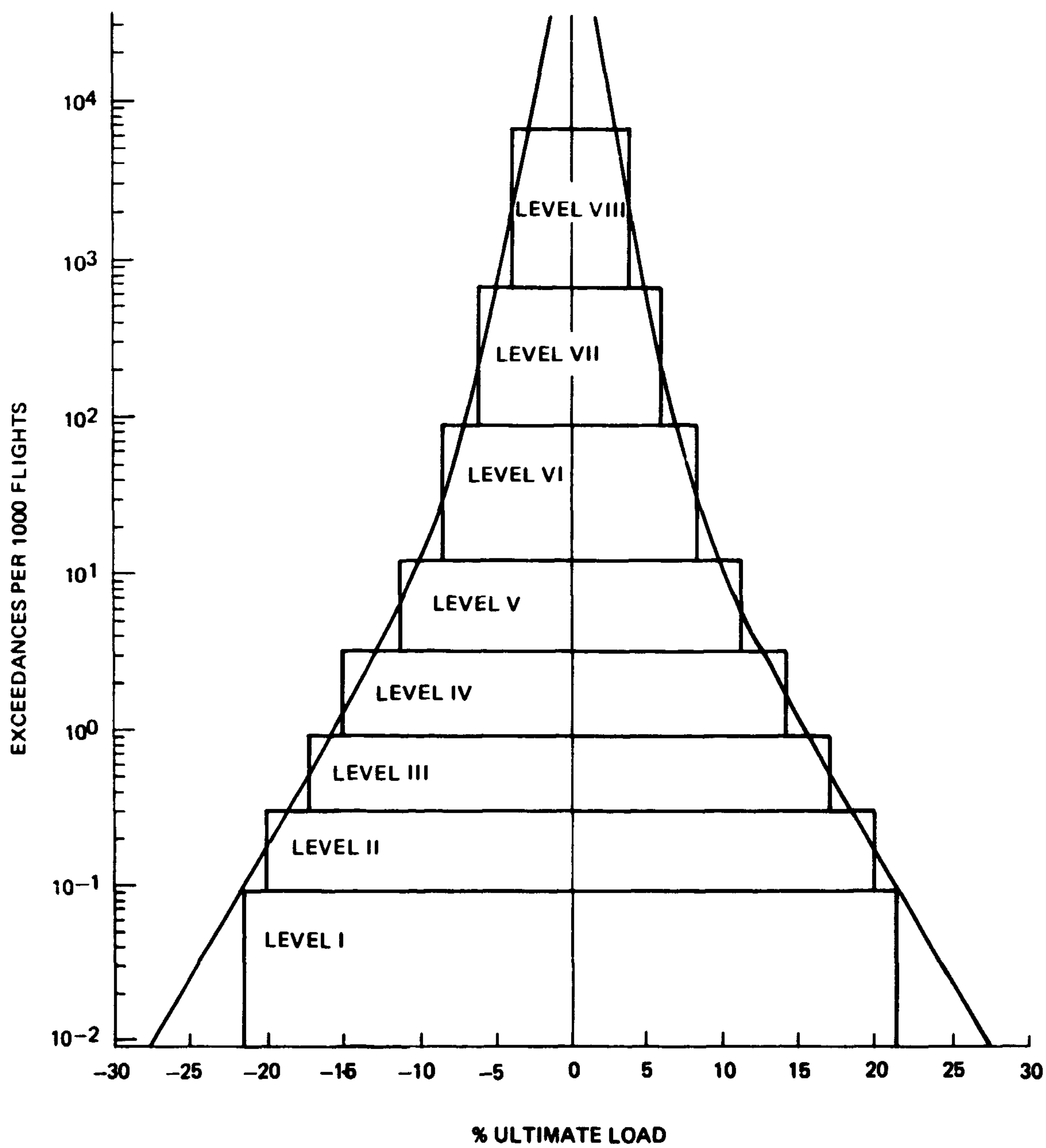

Figure 2-3. Gust Alternating Load Levels

Table 2-1. Alternating Load Occurrence Summary

Load type	Load level	Load cycle occurrences in 10,000 flights			
		Climb	Cruise	Descent	Total
Gust	VIII	8,797	40,548	13,966	63,311
	VII	718	3,955	1,128	5,801
	VI	87	575	138	800
	V	8	74	13	95
	IV	2	18	2	22
	III	1	5	1	7
	II	0	2	0	2
	I	0	1	0	1
Maneuver	VIII	11,040	55,722	14,896	81,658
	VII	2,152	10,122	2,699	14,973
	VI	426	1,875	497	2,798
	V	85	350	92	527
	IV	17	67	17	101
	III	3	12	3	18
	II	1	2	1	4
	I	0	1	0	1

types were defined to produce an array in which each succeeding flight includes a larger load level. The resulting frequency and cyclic load content of the eight flight types are shown in Table 2-2.

The distribution of gust and maneuver loads between climb, cruise, and descent test phases in each test flight type was made to match the overall distribution for 10,000 flights shown in Table 2-1. The resulting gust and maneuver load allocation for these three test phases is shown in Table 2-3. The sequence of flight types in the 10,000-flight block will be controlled, to result in a uniform distribution of flight types.

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Table 2-2. Flight Type Definition

Flight type / [1]	Number of gust load cycles at 8 amplitude levels I	II	III	IV	V	VI	VII	VIII	Number of maneuver load cycles at 8 amplitude levels I	II	III	IV	V	VI	VII	VIII	Number of load points in one flight
A / 1	1	1	.	2	6	14	112	766	1	3	5	2	7	3	2	3	1,866
B / 1		1	1	2	6	10	91	655		1	3	3	7	2	2	2	1,578
C / 5			1	1	2	2	39	468			2	8	7	3	1	5	1,084
D / 14				1	1	2	14	166				4	12	8	6	7	448
E / 62					1	2	4	73					5	13	10	15	252
F / 620						1	3	15						3	8	10	86
G / 3,100							1	6							3	8	42
H / 6,200								4								8	30

[1] Number of flights in a 10,000-flight block

2.2 DESIGN STATUS

The design effort is proceeding with the detailed design of all graphite/epoxy components, consisting of ribs, spars, skin panels, and trailing-edge beams. Detailed design of interfacing nongraphite/epoxy components, such as the inboard gap covers, leading-edge ribs, and thermal expansion compensating linkage, is also progressing as scheduled.

2.2.1 Stabilizer Box Assembly Provision

The stabilizer box will be assembled with titanium mechanical fasteners. Titanium Hi-Lok fasteners are used whenever possible, as these fasteners have lower installation cost and weight than other competing fastener systems. These Hi-Lok fasteners will be used generally with corrosion resistant steel (cres) collars placed over cres washers.

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Table 2-3. Alternating Load Allocation for Climb, Cruise, and Descent Test Phases

Flight type / [1]	Climb gust: Number of load cycles at 8 amplitude levels								Cruise gust: Number of load cycles at 8 amplitude levels								Descent gust: Number of load cycles at 8 amplitude levels							
	I	II	III	IV	V	VI	VII	VIII	I	II	III	IV	V	VI	VII	VIII	I	II	III	IV	V	VI	VII	VIII
A / 1			1	1	1	2	14	109	1	1	0	0	4	9	73	489			1	1	1	3	25	168
B / 1				1	1	2	11	91		1	1	0	4	6	60	419				1	1	2	20	145
C / 5					1	1	3	65			1	1	1	0	28	300					0	1	8	103
D / 14						1	0	24				1	0	0	10	107					1	1	4	35
E / 62						1	1	8					1	0	1	51						1	2	14
F / 620							1	2						1	1	9							1	4
G / 3,100								0							1	5								1
H / 6,200								1								2								1

Flight type / [1]	Climb maneuver: Number of load cycles at 8 amplitude levels								Cruise maneuver: Number of load cycles at 8 amplitude levels								Descent maneuver: Number of load cycles at 8 amplitude levels							
	I	II	III	IV	V	VI	VII	VIII	I	II	III	IV	V	VI	VII	VIII	I	II	III	IV	V	VI	VII	VIII
A / 1		1	2	1	3	1	1	0	1	1	1	0	1	1	1	2		1	2	1	3	1	0	1
B / 1			1	1	2	1	1	0		1	1	1	1	0	1	1			1	1	4	1	0	1
C / 5				3	2	2	0	2			2	2	2	0	0	3				3	3	1	1	0
D / 14					5	3	3	4				4	2	1	1	2					5	4	2	1
E / 62						6	4	7					5	0	3	8						7	3	0
F / 620							3	2						3	1	4							4	4
G / 3,100								1							3	7								0
H / 6,200								1								5								2

[1] Number of flights in a 10,000-flight block

Titanium bolts, with CRES nuts or nutplates, will be used whenever internal access is limited for Hi-Lok installation tools.

In assembling the stabilizer box, the front and rear spars will be joined initially to the ribs. Hi-Lok fasteners are generally used to join ribs to spars.

The upper panel will be fitted to the substructure (spars and ribs), using shims where they are required for proper fit, and then fastener holes will be drilled to join the skin to the substructure. Next, the panel will be removed, and nutplates will be installed on the substructure where internal access to the stabilizer box is limited.

The lower panel will be fitted next, shimmed, and installed with Hi-Lok fasteners.

The upper panel is next refitted and installed with bolts. These bolts, located on the outboard three-fourths of the stabilizer, will be installed using nutplates. The remaining bolts on the inboard area will be installed with nuts and washers. Accessibility to these nuts will be provided through access holes in the spars and inboard closure rib.

2.2.2 Stabilizer Box Access Provision

Inspection and manufacturing access provision in the stabilizer box is shown in Figure 2-4.

The 5.08-cm (2-in) diameter holes on the spars are for visual inspection of the interior only. The 10.16-cm (4-in) diameter access holes are used primarily for inspection, but they are also used for manufacturing access.

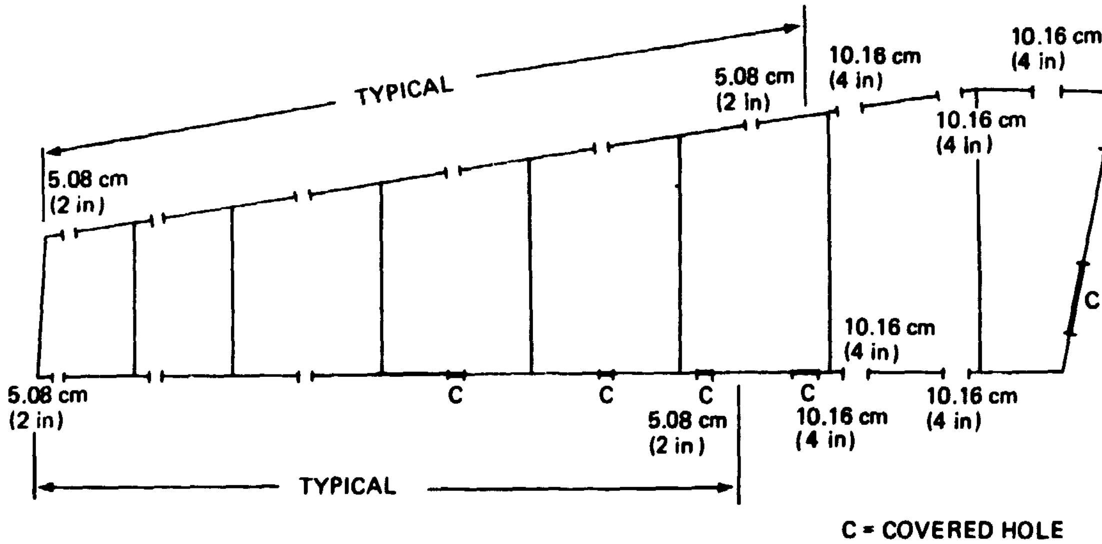

Figure 2-4. Access and Inspection Provision

The large access holes provided on the inboard closure rib can be used for visual inspection of the structurally important details at the inboard areas of the spars.

The holes in the rear spar at the elevator balance panel bays are provided with covers, as indicated in Figure 2-4. These covers prevent the unregulated air pressure of the stabilizer box interior from disturbing elevator balance pressures in the balance bays. A covered inspection hole is illustrated in Figure 2-5.

2.2.3 Corrosion Protection

Corrosion protection will be provided to each aluminum component near graphite/epoxy structure, to minimize the possibility of galvanic corrosion.

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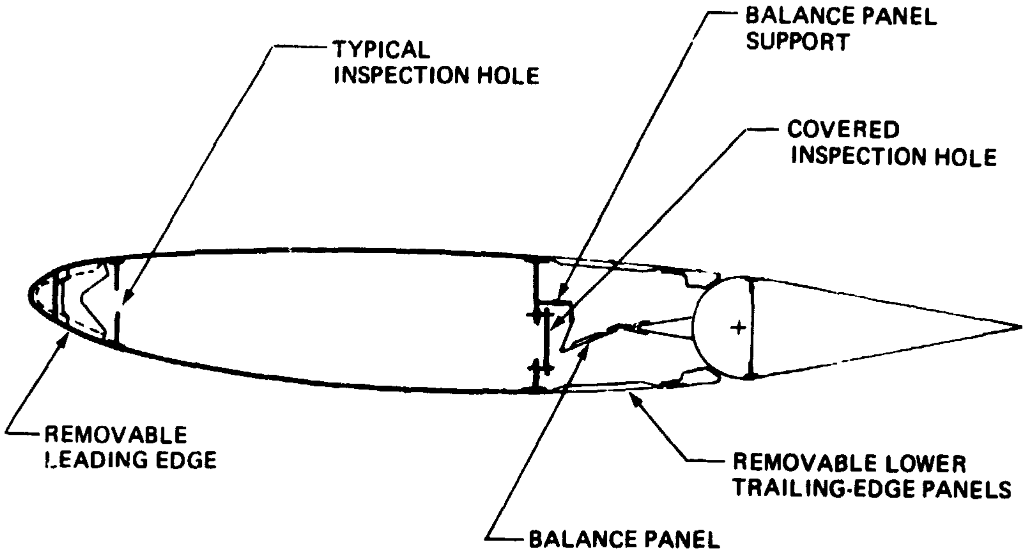

Figure 2-5. Inspection Holes in Spars

The general concept is to isolate the graphite/epoxy near the aluminum by careful application of finishes and coverings of the graphite/epoxy.

Aluminum components will be anodized or alodine treated, primed, and enameled.

The graphite/epoxy surface that interfaces with aluminum will be covered with a ply of fiberglass cocured with the graphite/epoxy.

All graphite/epoxy surfaces near aluminum, including cut edges not provided with cocured fiberglass ply, will be primed and enameled. An exception is where Tedlar film can be applied to the graphite/epoxy layup during cure. Tedlar film is preferred over primer and enamel on the graphite/epoxy surfaces near aluminum, because Tedlar is lighter, and the cost of application is less than that of paint.

Aluminum components will be joined to the graphite/epoxy with faying surface sealant. Fasteners joining aluminum and graphite/epoxy will be installed with wet sealant.

Where the aluminum component is a removable part, the faying surface sealant will not be used. Fastener hole and countersunk surfaces will be alodine-treated, primed, and enameled.

The corrosion protection system used is identical to that used on the 727 advanced composites elevator, being developed under NASA Contract NAS1-14952.

2.2.4 Skin Panel Stiffener Runout Detail

Panel stiffener inboard end runout detail has been changed as shown in Figure 2-6.

The previous design required locating the ends of the stiffener plies precisely on the skin layup, to coordinate with the edge of the inboard closure rib flange. The new design does not have this requirement, as the stiffener plies extend under the rib to the trimmed edge of the panel.

A concern over the possibility that the end-load transfer from the stiffener to the skin, combined with the bending load from air pressure, could initiate stringer delamination contributed to the decision to change this detail.

Filler plies will have to be added between the stiffener plies under the rib. The extended stiffener plies and the filler plies add 0.068 kg (0.15 lb) to each skin panel.

2.2.5 Rib Corner Detail

The honeycomb rib design detail at the forward corners has been changed, as shown in Figure 2-7, to facilitate manufacture, based on experienced gained during fabrication of the verification hardware. The basic problem is that the graphite/epoxy material tends to "bunch-up" at the corners, resulting in thicker than desirable laminate in these areas. This thickness creates fit-up problems at the front spar where flat, well-matched interfaces are required.

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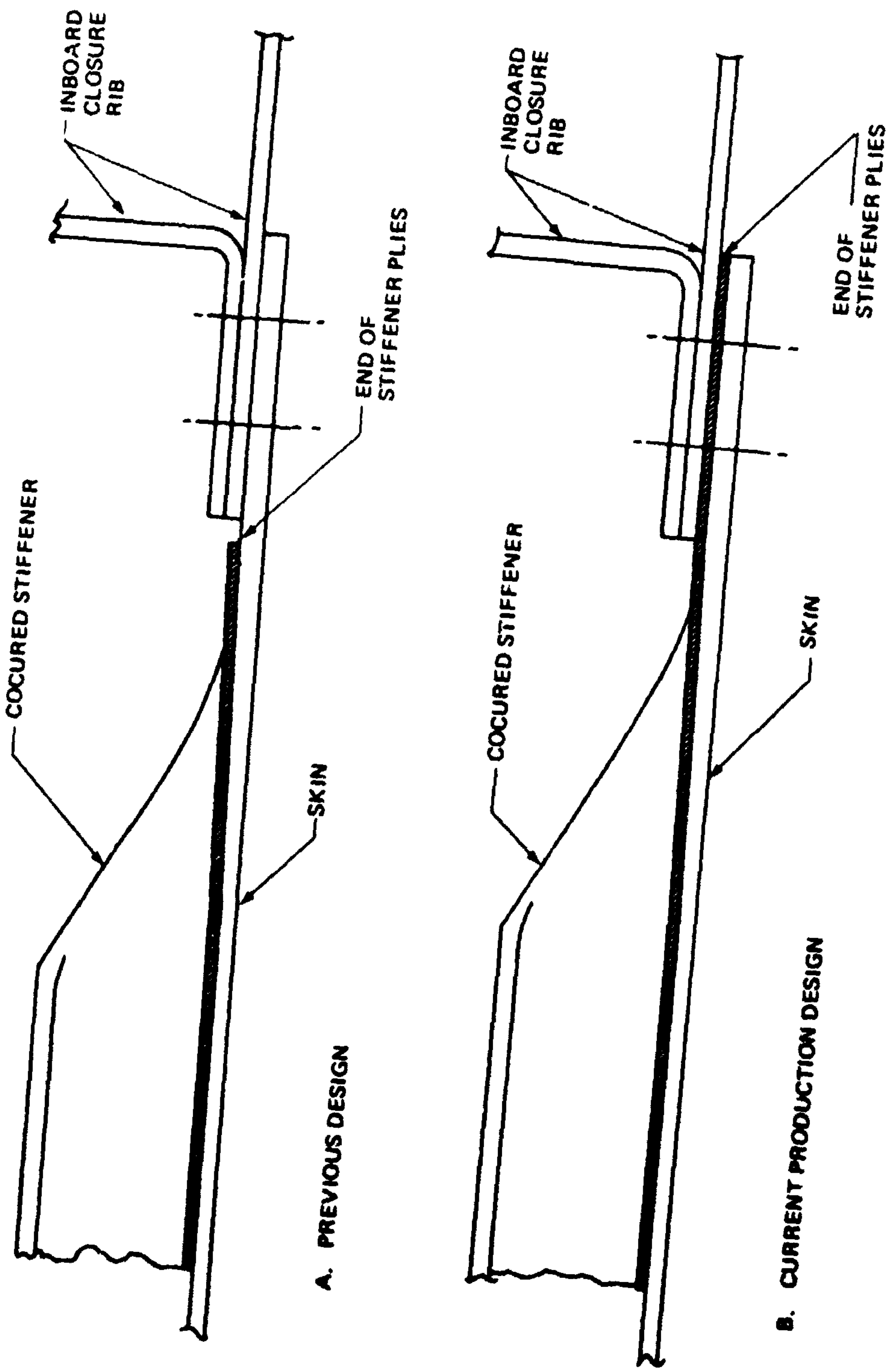

Figure 2-6. Stiffener Runout Details

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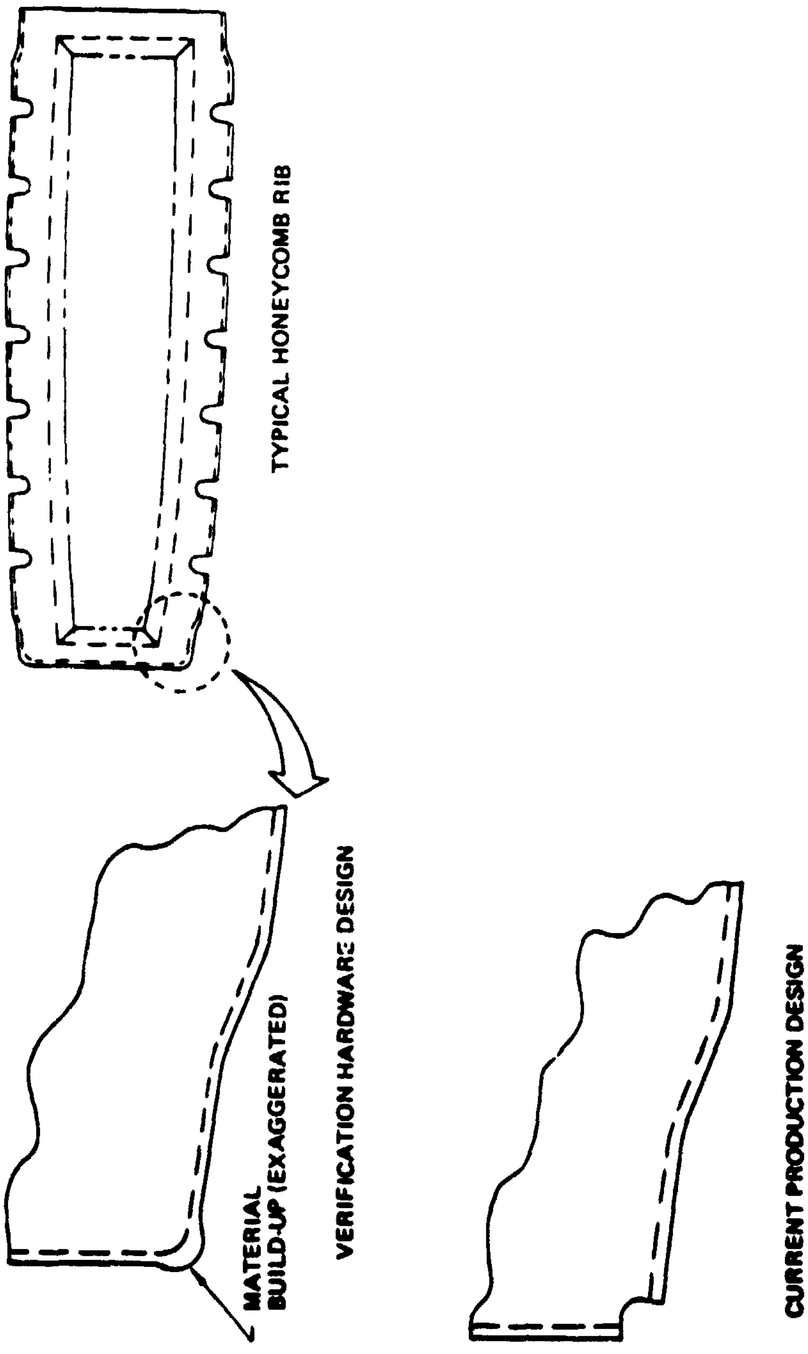

Figure 2-7. Honeycomb Rib Forward Corner Details

2.2.6 Production Drawing Preparation

The following drawings have been completed and released to the production shops:

65C17810	Rib Installation - Stabilizer Station 83.50
65C17811	Rib Installation - Stabilizer Station 111.10
65C17812	Rib Installation - Stabilizer Station 138.70
65C17818	Rib Installation - Outboard Closure
65C17825	Attach Angle - Inboard Closure Rib
65C17847	Gap Cover and Seal Installation
65C17860	Beam Installation - Trailing Edge
65C17861	Beam Assembly - Trailing Edge
69-69807	Tapered Filler
69-69808	Attach Fitting - Inboard Closure Rib
69-69809	Attach Fitting - Inboard Closure Rib
69-69810	Attach Fitting - Inboard Closure Rib
69-69811	Attach Fitting - Inboard Closure Rib
69-69812	Attach Fitting - Inboard Closure Rib

The following drawings are essentially complete. Final checking is being conducted prior to approval and release:

65C17819	Rib - Inboard Closure
65C17831	Front Spar Channel Assembly
65C17832	Rib Installation - Leading Edge Station 56.01
65C17833	Rib Installation - Leading Edge Station 86.66
65C17834	Rib Installation - Leading Edge Station 69.93
65C17837	Rib Installation - Leading Edge Station 78.29
65C17841	Rear Spar Channel Assembly
65C17845	Leading Edge Installation - Fixed
65C17857	Bellcrank - Thermal Expansion Adjusting

2.3 ANALYSIS

An internal loads analysis is being performed to determine load levels in all elements of the stabilizer. A finite element structural model, using the Boeing ATLAS program, is being used for this analysis. The structural box definition includes both skins, front and rear spars, inspar ribs, trailing-edge panels, and the elevator support ribs and hinge supports. The elevator is simulated by a beam with the same effective bending stiffness. The structural model is supported by flexible members, to simulate the stiffness of the stabilizer center section. This mounting procedure ensures correct distribution of bending and shear between the front and rear spars. The finite element model will be analyzed, and several resize iterations will be performed to refine the initial element sizing. All design load cases will be analyzed for a complete check after the model has been refined.

The advanced composites stabilizer has been analyzed, and the internal loads have been generated by the ATLAS model. This information has been used to size the inboard portion of the stabilizer structure for the stub box test component. Examples of the ATLAS model internal loads and the resulting structural details are presented in Reference 6.

During this reporting period, a thermal analysis for establishing the maximum test temperature was initiated. The overall thermal model used to describe the boundary conditions is presented in Figure 2-8. The conditions and the assumptions used for the steady-state condition are as follows:

- Zero wind velocity
- Effective sky temperature = $-17^{o}C$ ($0^{o}F$)
- Ambient air temperature = $45^{o}C$ ($113^{o}F$) (This value would not be exceeded 95% of the time, based on a survey of worldwide airport conditions)
- Asphalt temperature = $59^{o}C$ ($138^{o}F$) (Based on a recent worldwide survey conducted by Boeing)

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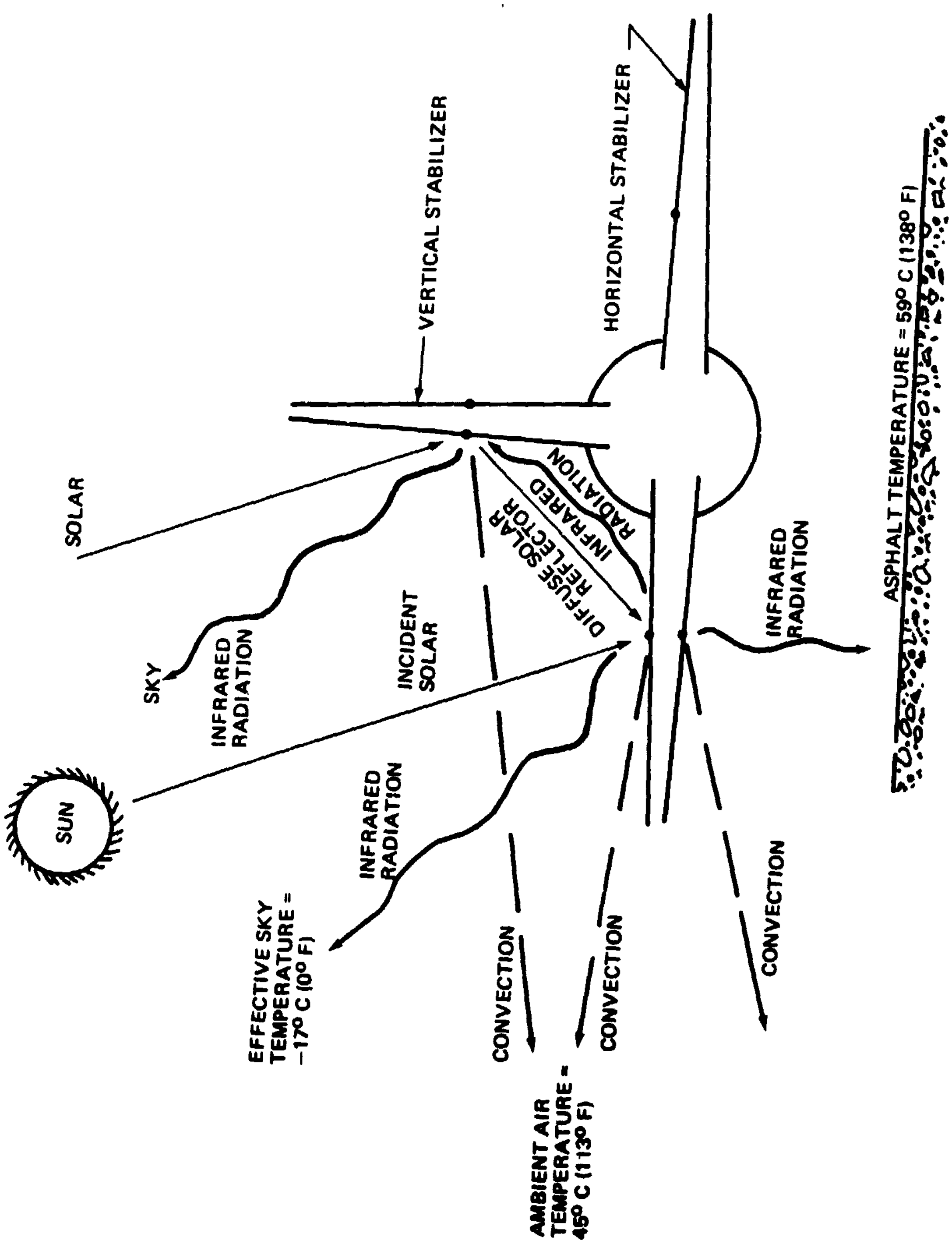

Figure 2-8. 737 Horizontal Stabilizer Thermal Model

- Vertical tail is painted white (Solar absorptivity α_S = 0.252 and infrared emissivity ϵ_{IR} = 0.910)
- Sun angle from the vertical was fixed at 15°
- Infrared emissivity of uncoated graphite/epoxy interior surfaces ϵ_{IR} = 0.70

The values of thermal conductivity and specific heat used in the program are presented in Table 2-4. These values have been taken from Reference 7.

Table 2-4. Values of Thermal Conductivity and Specific Heat for Graphite/Epoxy Advanced Composites Laminate

Physical quantity [1]	Units	Value
Thermal conductivity along the fiber	W/M-K Btu/hr-ft-°F	15.00 8.67
Thermal conductivity across the fiber	W/m-K Btu/hr-ft-°F	1.50 0.87
Specific heat	Cal/gm-K	0.25

[1] Value at 366 K (200° F)

The stringer and the skin panel used in the thermal model are shown in Figure 2-9. Those sections shown in Figure 2-9 represent the thickest gages that occur on the stabilizer skin panels. Thinner gages than those shown in Figure 2-9 were also analyzed; but the analysis results indicated that the thicker gages attained the higher temperatures in both the steady-state and transient cases. The steady-state temperatures for various light and dark colored paints are shown in Table 2-5. These results indicate that the maximum steady-state temperatures are highest with paint systems that have a high solar absorptivity and a low infrared emissivity.

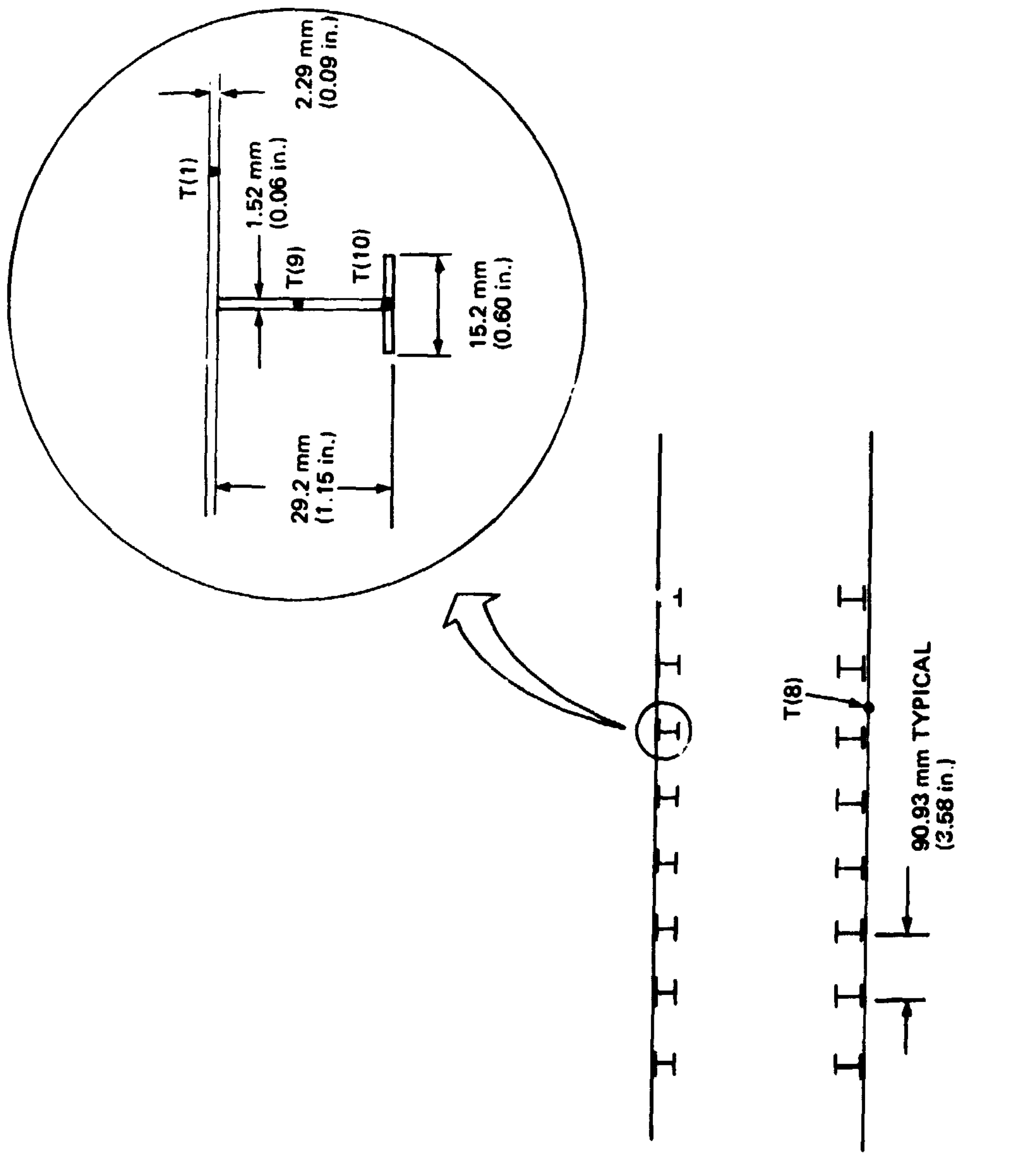

Figure 2-9. Stringer Detail for Thermal Model

Table 2-5. 737 Stabilizer Steady-State Temperatures

Paint color	Surface texture	Solar absorptivity α_s	Infrared emissivity ϵ_{IR}	Top surface temperature T (1) °C (°F)	Bottom skin temperature T(8) °C (°F)	Stringer web T(9) °C (°F)	Stringer flange T(10) °C (°F)
7067 White	Clean	0.252	0.910	52.2 (126)	55.0 (131)	52.8 (127)	52.8 (127)
	Aged*	0.265	0.935	52.2 (126)	55.0 (131)	52.8 (127)	52.8 (127)
702 White	Clean	0.316	0.920	57.8 (136)	56.7 (134)	57.8 (136)	57.2 (135)
	Aged	0.323	0.955	57.2 (135)	56.1 (133)	57.2 (135)	56.7 (134)
707 Grey	Clean	0.560	0.915	74.4 (166)	61.1 (142)	73.9 (165)	72.8 (163)
	Aged	0.554	0.945	74.4 (166)	61.1 (142)	73.3 (164)	72.8 (163)
7025 Grey	Clean	0.742	0.925	86.1 (187)	65.6 (150)	85.0 (185)	83.9 (183)
	Aged	0.727	0.960	82.2 (180)	63.9 (147)	81.7 (179)	80.6 (177)
5109 Blue	Clean	0.900	0.720	99.4 (211)	71.1 (160)	98.3 (209)	96.7 (206)
701 Black	Clean	0.950	0.680	105.0 (221)	73.9 (165)	103.3 (218)	101.7 (215)

*Aging was simulated by surface roughening with sandpaper

The conditions and assumptions used for the transient cases are defined as follows:

- Four-min taxi run, with a constant relative wind velocity of 33.8 km/hr (20 knots), followed by constant acceleration to 321 km/hr (190 knots) in 1.2 min. This point has been selected as the earliest possible time that the aircraft could be subjected to high design loads.

- The heat transfer coefficient as a function of velocity is shown in Figure 2-10.

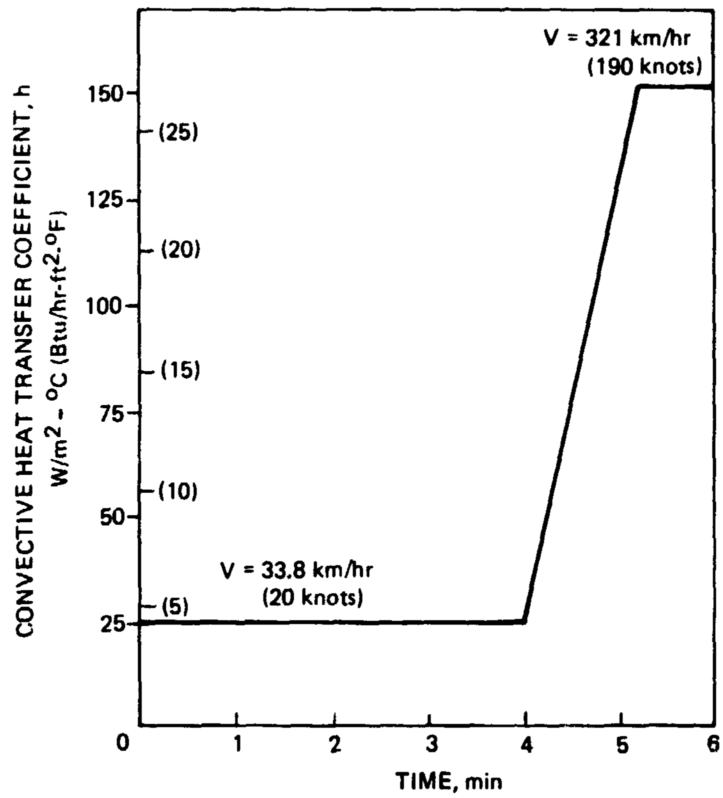

Figure 2-10. Heat Transfer Coefficient vs Velocity for Transient Analysis

Several transient thermal cases were analyzed, and the results were reviewed to determine the maximum temperatures that could be expected. Three of the most severe cases are presented in Figures 2-11 through 2-13. Results of a 4-min taxi case for a gray, blue, and black painted surface are presented in Figures 2-11, 2-12, and 2-13.

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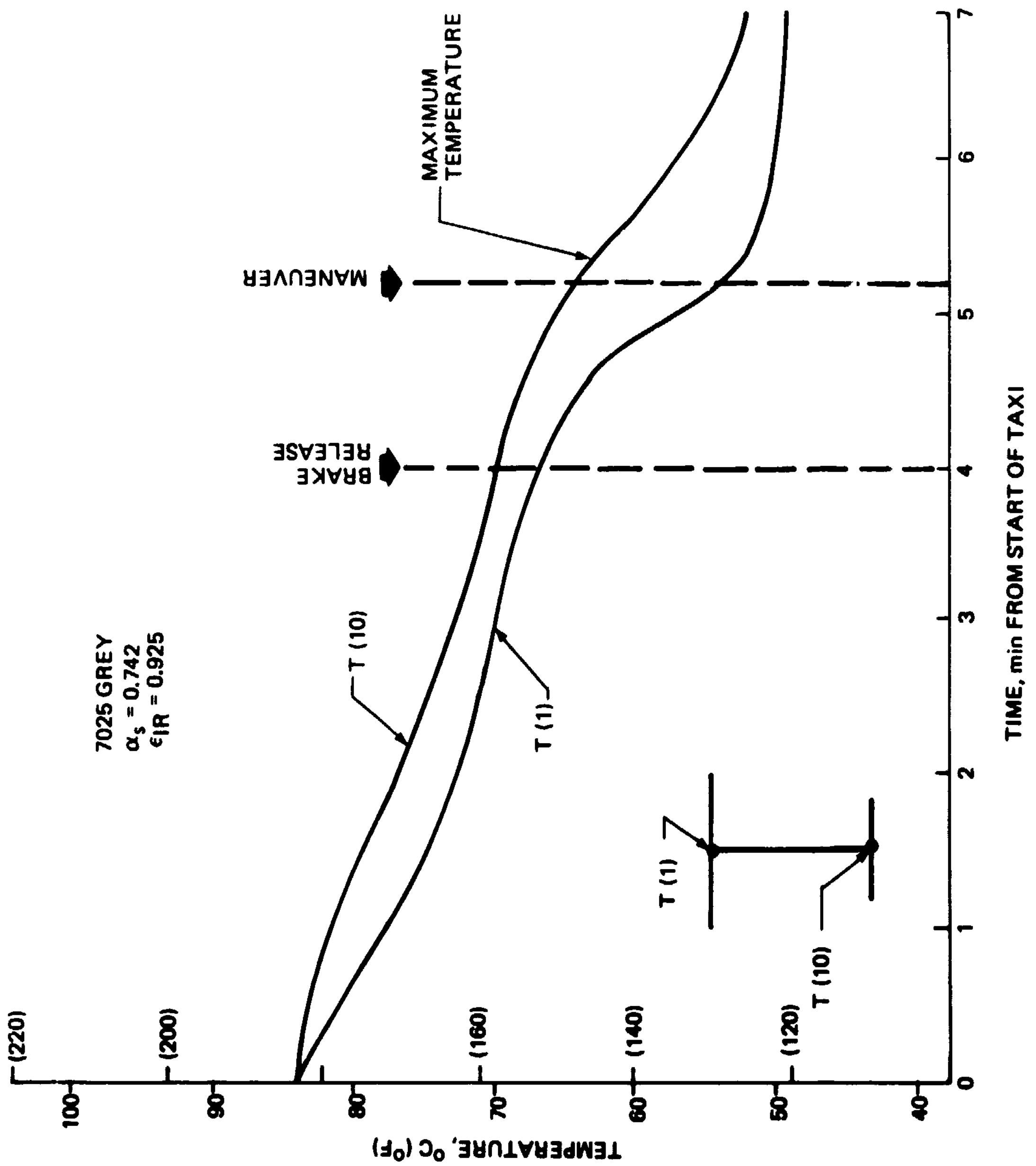

Figure 2-11. Transient Thermal Response

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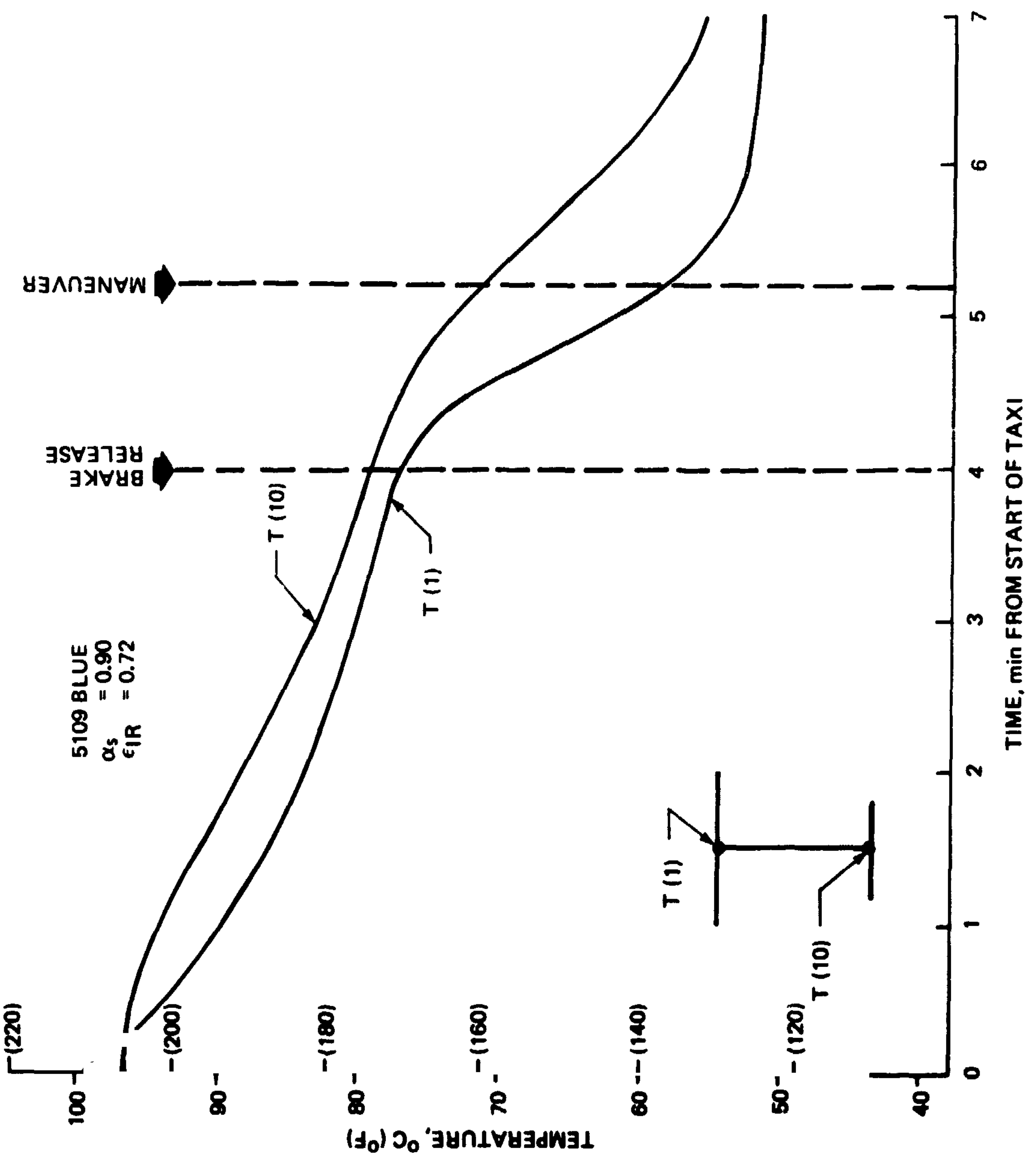

Figure 2-12. Transient Thermal Response

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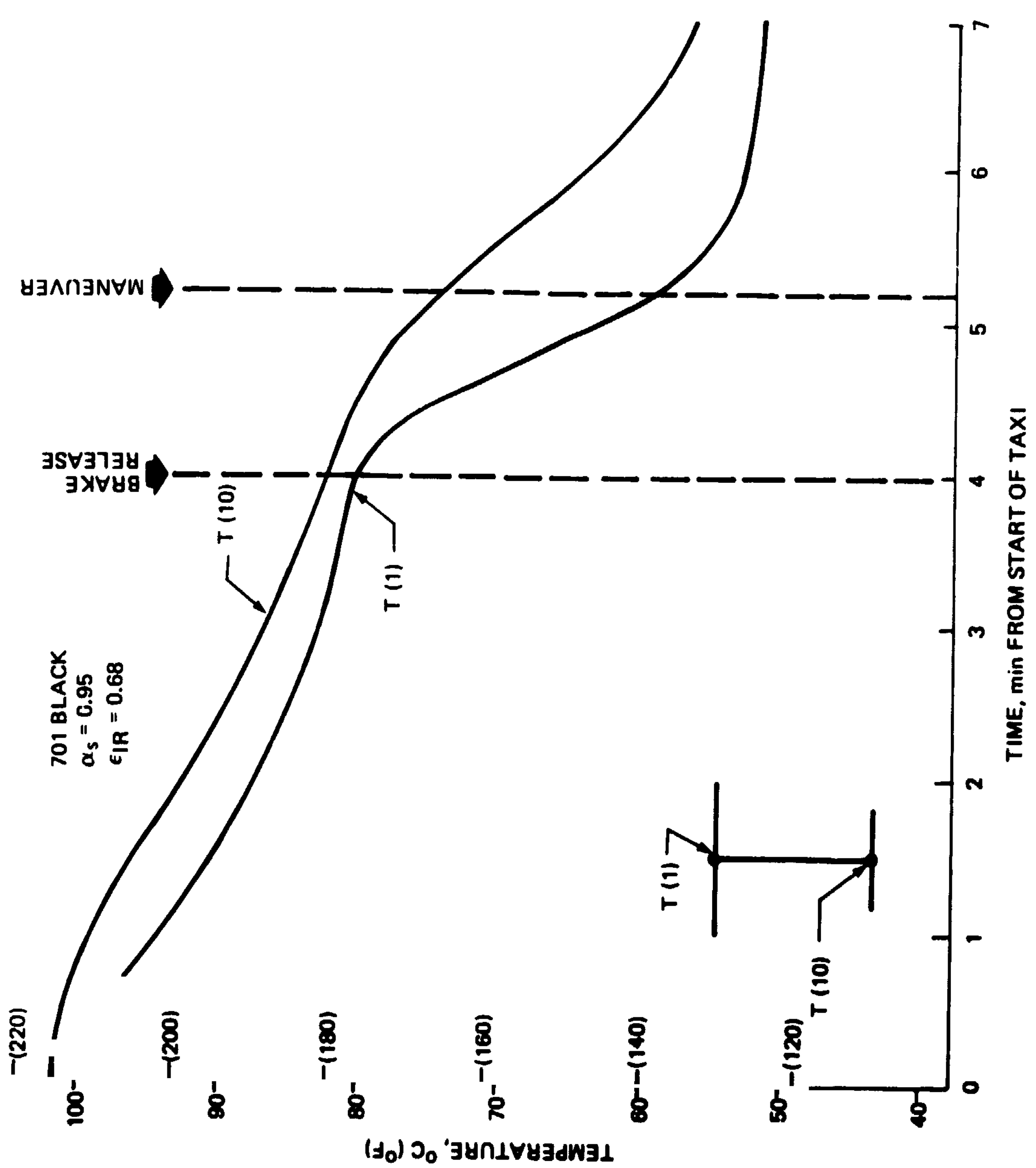

Figure 2-13. Transient Thermal Response

Results of the transient thermal analysis are as follows:

- The stringer inner flange (T 10) does not cool as rapidly as the skin
- The paint system that attains the highest steady-state skin temperature also has the highest (T 10) stringer flange temperature at the end of the 1.2-min acceleration period
- For the 4-min taxi case for the black painted surface, the maximum temperature of the stringer inner flange (T 10) is 77°C (170°F)

Based on results of this thermal analysis, the coupons and the subcomponents in the ancillary test plan will be tested at 82°C (180°F). This test temperature was selected because it was a representative maximum temperature to cover the worst case of a dark-colored paint system.

The analysis task of establishing the stabilizer stiffness to meet stability/control and flutter requirements is still proceeding. Figures 2-14 and 2-15 present the most recent stiffness calculations, based on the stub box skin gages (Test No. 21). The bending stiffness (EI) curves have been modified from previously published curves, to reflect a refined analysis that accounts for material effectivity due to stabilizer sweep-back and shear-lag. These curves also present a comparison between the buckled aluminum and advanced composites skins. This information is presently being evaluated by the Stability/Control and Flutter Technology groups.

2.4 WEIGHT STATUS

The stabilizer skin panel weight distribution has been reevaluated by utilizing the stub box gages, Test No. 21, as being representative of the production inboard section. Outboard of this stub box, Stabilizer Station

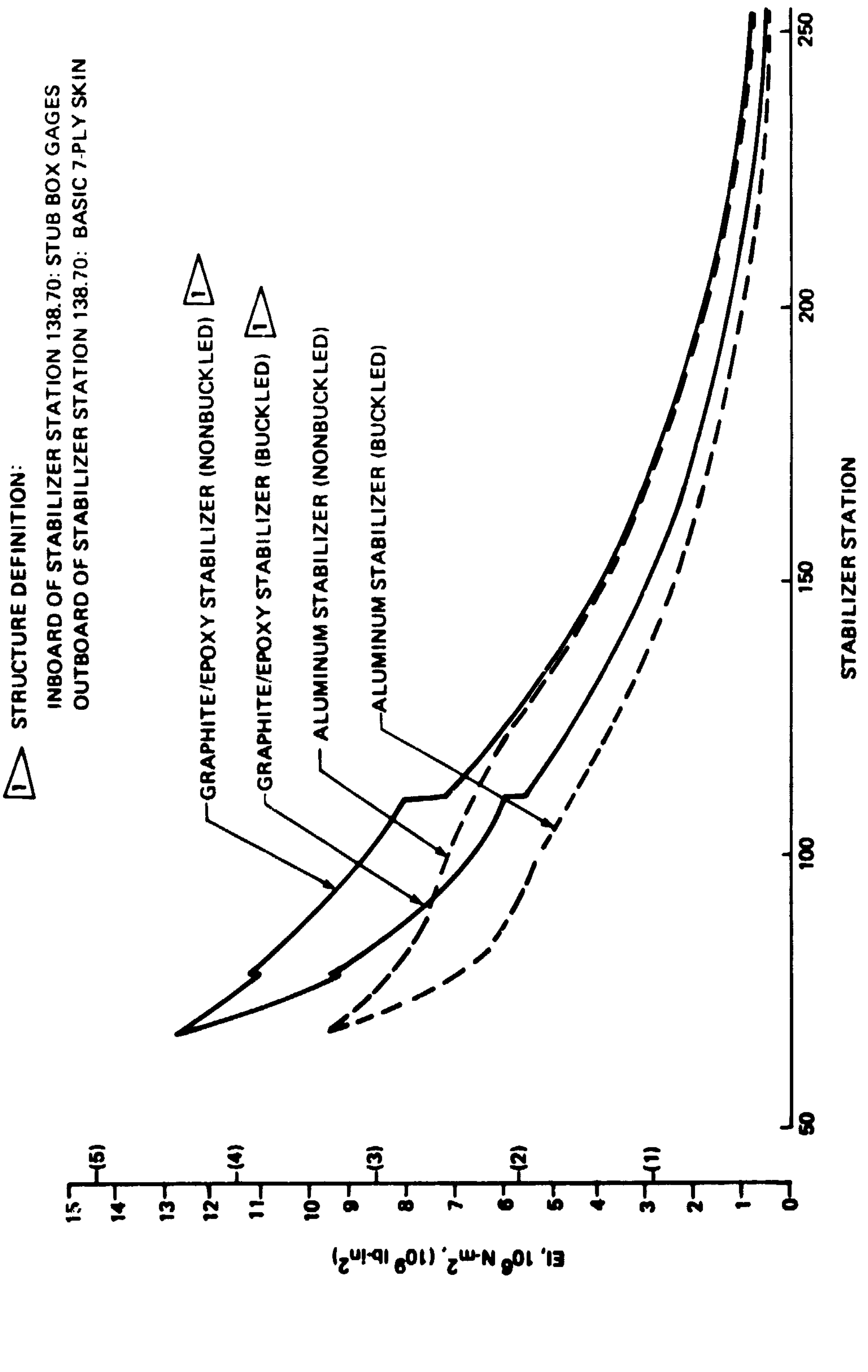

Figure 2-14. Stabilizer Bending Stiffness

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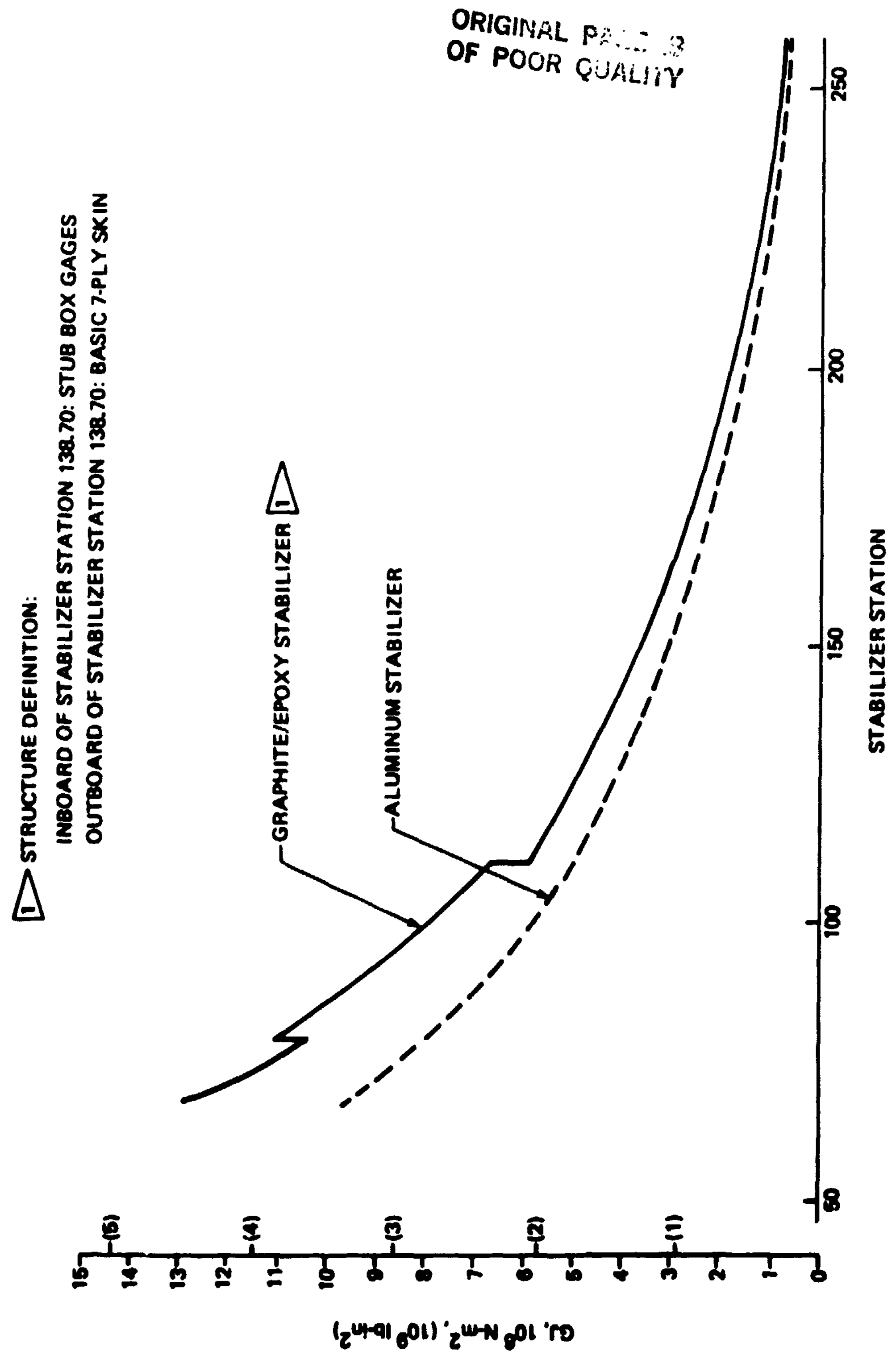

Figure 2-15. Stabilizer Torsional Stiffness

138.7C, the weight has been extrapolated using minimum gage. The evaluation results in an increase to the skin panel weights, and a stabilizer percentage weights savings decrease from 29% to 27%. See Table 2-6.

The stabilizer mass properties for flutter and vibration analysis was conducted using the above skin panel gages.

Evaluation of the stub box drawings is continuing, with 65% completion.

Table 2-6. Advanced Composites Horizontal Stabilizer Inspar Structure Weight Comparison—737
Previous report: Second quarterly (January 1978)

Item	① Aluminum baseline, kg (lb)/airplane	② Advanced composites system weights: Previous report, kg (lb)/airplane	③ Advanced composites system weights: Current report, kg (lb)/airplane	④ Advanced composites system weights: Δ Previous to current, kg (lb)/airplane ② — ③	⑤ Weight difference, kg (lb)/airplane ① — ③	⑥ % Weight difference
Front spar	31.3 (69.0)	20.2 (44.6)	20.2 (44.6)	0 (0)	−11.1 (−24.4)	−35
Rear spar	71.1 (156.8)	42.9 (94.5)	42.9 (94.5)	0 (0)	−28.3 (−62.3)	−40
Skins— upper △	36.2 (79.8)	31.0 (68.2)	34.5 (76.1)	+3.5 (+7.9)	− 1.7 (− 3.7)	− 5
lower △	36.2 (79.8)	33.6 (74.2)	36.3 (80.1)	+2.7 (+5.9)	+ 0.1 (+ 0.3)	+ 0
Ribs	60.9 (134.2)	30.3 (66.8)	30.3 (66.8)	0 (0)	−30.6 (−67.4)	−50
Corrosion protection	—	6.8 (15.0)	6.8 (15.0)	0 (0)	+ 6.8 (+15.0)	—
Lightning protection system	—	0.4 (1.0)	0.4 (1.0)	0 (0)	+ 0.4 (+ 1.0)	—
Access doors	0.7 (1.6)	0.9 (2.1)	0.9 (2.1)	0 (0)	+ 0.2 (+ 0.5)	+31
Total stabilizer inspar structure/airplane	236.4 (521.2)	166.1 (366.4)	172.3 (380.2)	+6.2 (+13.8)	−64.0 (−141.0)	−27
Stabilizer trailing-edge/ elevator interface thermal expansion provision	—	+7.0 (+15.5)	+7.0 (+15.5)	+7.0 (+15.5)	+ 7.0 (+15.5)	—

△ Skin panel weight distribution refined to reflect stub box (test No. 21) skin gages

SECTION 3.0

DEVELOPMENT TEST PLAN AND STATUS

3.1 ANCILLARY TEST PROGRAM

During this reporting period, the following test programs have been defined, and the drawings have been released. The drawings are presented in Appendix A.

- Test No. 10 - Skin Panels - Drawing 65C17773
- Test No. 20 - Sonic Box - Drawing 65C17792
- Test No. 22 - Discontinuous Laminate - Drawing 65C17980
- Test No. 24 - Pressure/Shear Skin Rib Joint - Drawing 65C17981

The ancillary test program has been revised to reflect the completion of the Test No. 10 drawings, and to include Test No. 22 and Test No. 24. The revised test program is presented in Figures 3-1 through 3-7. The production verification hardware test program (see Figure 3-6) has been assigned as Test No. 25. The ancillary test program schedule is shown in Figure 3-8.

During this reporting period, 24 bolted joint specimens of Test No. 5 were tested. The test specimens are defined in Figures 3-9 and 3-10. The test results are presented in Tables 3-1 and 3-2. The net area stress and bearing stress that existed at the time of failure is plotted in Figures 3-11 and 3-12. These test results show that design bearing stresses are significantly influenced by fastener spacing. This series of tests also included wet testing at room temperature. These specimens are presently undergoing moisture conditioning, and will be tested when the required moisture content has been attained.

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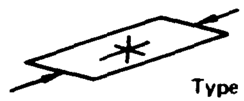
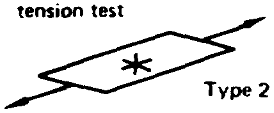
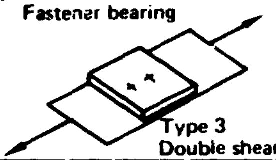
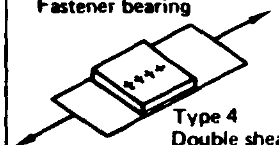
Test No.	Drawing No.	Specimen configuration	Cloth laminate, deg	Size, mm (in.)	Condition Wet (W) Dry (D)	Number of specimens and test temperatures: Room temperature	-54°C	+82°C	Data	Instrumentation	Purpose	Remarks
Test No. 1	65C17768	Impact defect compression test  Type 1	0/±45$_2$/90	406 x 76 (16 x 3)	W	12		6	Load/strain	Extensometer	Effect of stress concentration	• Two thicknesses • Two impact levels
					D	12	6	6				
			0/±45/90		W	12		6				
					D	12	6	6				
Test No. 1	65C17768	Impact defect tension test  Type 2	0/±45/90	406 x 76 (16 x 3)	W	12		6	Load/strain	Extensometer	Effect of stress concentration	• Two thicknesses • Two impact levels
					D	12	6	6				
Test No. 1	65C17768	Fastener bearing  Type 3 Double shear	0/±45$_2$/90	381 x 14.2 (15 x 0.56) to 381 x 31.75 (15 x 1.25)	W	12	3	12	Failure load and mode of failure		Bearing strength	• Two sizes of fastener • Two W/Ds
					D	12	12	3				
Test No. 1	65C17768	Fastener bearing  Type 4 Double shear	0/±45/90	318 x 23.9 (15 x 0.94) to 381 x 44.45 (15 x 1.75)	W	12			Failure load and mode of failure		Bearing strength	• Two sizes of fastener • Two W/Ds
					D	12	12	12				

Note: Wet condition denotes 1.1% moisture content achieved by conditioning in a 100% relative humidity chamber at 60°C

Figure 3-1. Material Allowables Testing—Mechanical Properties

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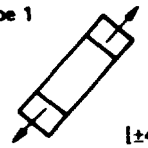
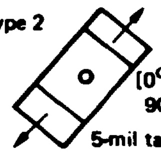
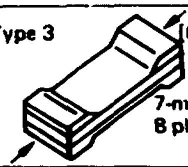
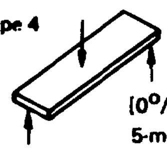
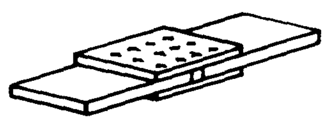
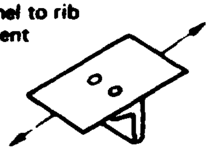
Test No.	Drawing No.	Specimen configuration	Size, mm (in.)	Exposure conditions	Exposure time (months and hours) 0	6	12	18	24	36	Type of test after exposure (test at room temperature)	Exposure conditions
					0	4,380	8,760	13,140	17,520	26,280		
Test No. 4	65C17703	Type 1  [±45°] 8T 5-mil tape	305 x 25.4 (12 x 1)	I	5		5		5	5	Static tension	I Laboratory shelf exposure II Outdoor rack exposure, Strained during exposure III Webber chamber temperature, humidity, and pressure cycling. Strained during exposure. IV 100% relative humidity at 60°C [1] Three of each series of six specimens will be initially fatigue cycled to equivalent flight cycles corresponding to scheduled calendar time of exposure.
				II			5		5	5		
				III						5		
				IV			5		5			
Test No. 4	65C17703	Type 2  [0°/±45°/90°] 2S 5-mil tape [1]	381 x 38.1 (15 x 1.5)	I	3		6		6	6	Fatigue test to failure R = -1.0	
				II			6		6	6		
				IV			6		6	6		
Test No. 4	65C17703	Type 3  [0°/90°/±45°] 7-mil fabric 8 plies	305 x 25.4 (12 x 1)	I	5		5		5	5	Static compression	
				II			5		5	5		
				III						5		
				IV			5		5			
Test No. 4	65C17703	Type 4  [0°/90°] $\overline{8S}$ 5-mil tape	15.2 x 6.35 (0.6 x 0.25)	I	5		5		5	5	Static interlaminar shear test	
				II			5		5	5		
				III						5		
				IV			5		5			

Figure 3-2. Long-Term Environmental Assessment Test Plan

Test No.	Drawing No.	Specimen	Size, mm (in.)	Configuration	Condition Wet (W) Dry (D)	Test temperature and number of tests: Room temperature	-54°C	+82°C	Data	Remarks
Test No. 5	65C17769	Mechanical joint box fastener pattern  Laminate [0/±45₂/90]	533 x 71 (21 x 28) to 813 x 34 (32 x 5.25)	-1 to -6	D	12	–	–	Static tension failure load and mode	• Two fastener sizes • Two W/Ds
					W	12	–	–		
					D	6	–	–	Static compression failure load and mode	
					D	6	–	–	Fatigue life [2]	
Test No. 5	65C17769	Mechanical joint staggered fastener pattern  Laminate [0/±45₂/90]	432 x 43 (17 x 1.7) to 660 x 95 (26 x 3.75)	-1 to -4	W	12	–	–	Static tension failure load and mode	• Two fastener sizes • Two W/Ds
					D	12	–	–		
					D	6	–	–	Fatigue life [2]	
Test No. 9	65C17786	Skin panel to rib attachment 	508 x 152 (20 x 6)	-1	D	3	3	–	Failure load and mode of failure	Static strength
					W	3	–	3		
				-2	D	3	–	–		
				-1	D	3+3 [1]	3	–	Fatigue life Load and cycles to failure [2]	One life spectrum fatigue test followed by static test to failure
					W	3	–	3		

Note: Wet condition denotes 1.1% moisture content achieved by conditioning in a 100% relative humidity chamber at 60°C

[1] Two life spectra fatigue tests followed by static test to failure

[2] Damage growth rates measured

Figure 3-3. Design Development Structural Element Test Plan

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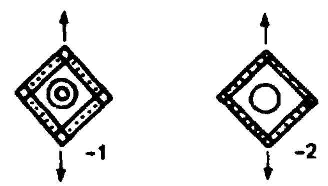
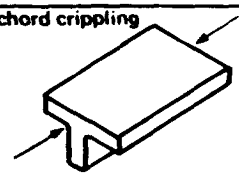
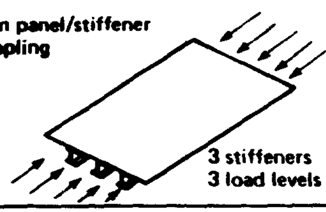
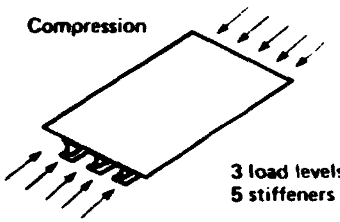
Test No.	Drawing No.	Specimen	Size, mm (in.)	Configuration	Condition Wet (W) Dry (D)	Test temperature and number of tests: Room temperature	-54°C	+82°C	Data	Remarks
Test No. 11	65C17789	Spar shear web  -1 Reinforced hole; -2 Unreinforced hole	305 x 305 (12 x 12)	-1	D	3	3	–	Failure load and mode of failure stiffness measure-ments	Static strength
					W	3	–	3		
				-2	D	3	3	–		
					W	3	–	3		
Test No. 7	65C17791	Spar chord crippling 	356 (14)	Front spar chord	D	3	3	–	Failure load and mode of failure	Static strength
					W	3	–	3		
				Rear spar chord	D	3	–	–		
					W	3	–	3		
Test No. 10	65C17773	Skin panel/stiffener crippling  3 stiffeners 3 load levels	305 x 305 (12 x 12)	-1	D	3	–	–	Failure load and mode of failure	Static strength
				-2	D	3	–	–		
				-3	D	3	–	–		

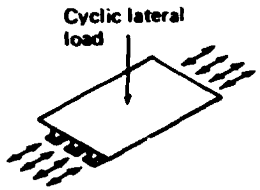
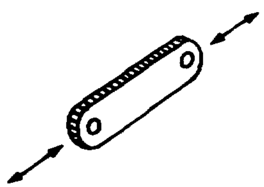
Note: Wet condition denotes 1.1% moisture content achieved by conditioning in a 100% relative humidity chamber at 60°C

Figure 3-3. Design Development Structural Element Test Plan (Concl)

Test No.	Drawing No.	Specimen	Size, mm (in.)	Config-uration	Condition Wet (W) Dry (D)	Test temperature and number of tests: Room temp-erature	Test temperature and number of tests: -54°C	Test temperature and number of tests: +82°C	Data	Load condition
Test No. 10	65C17773	Stiffened skin panel Compression  3 load levels 5 stiffeners	442 x 1400 (17.4 x 55)	-1	D	3 +1 [1] +2 [2]	1+1 [2]	–	Failure load and mode of failure	Static strength [1] Panels used for lightning strike protection testing [2] Defect or damaged part
					W	2+1 [2]	–	1+1 [2]		
				-2	D	1	–	–		
				-3		1	–	–		
Test No. 10	65C17773	Stiffened skin panel Shear loading  3 load levels 7 stiffeners	762 x 762 (30 x 30)	-1	D	3 +1 [1] +2 [2]	1+1 [2]	–		
					W	2+1 [2]	–	1+1 [2]		
				-2	D	1	–	–		
				-3		1	–	–		
Test No. 10	65C17773	Stiffened skin panel Compression and shear  2 load levels 7 stiffeners	762 x 7?? (30 x 30)	-1	D	3+2 [2]	–	–		
					W		–	1		
				-2	D	1	–	–		

Note: Wet condition denotes 1.1% moisture content achieved by conditi.. ning in a 100% relative humidity chamber at 60°C

Figure 3-4. Stabilizer Subcomponent Test Plan

Test No.	Drawing No.	Specimen	Size, mm (in.)	Configuration	Condition Wet (W) Dry (D)	Test temperature and number of tests: Room temperature	-54°C	+82°C	Data	Load condition
Test No. 10	65C17773	Stiffened skin panel—fatigue  Cyclic lateral load; 4 load levels, 4 stiffeners	1321 x 345 (52 x 13.6)	-1 ②	D	2	1	—	Fatigue life-load and cycles to failure	
					W	1	—	1		
				-2 ②	D	1	—	—		
				-3 ②						
				-4 ③	D	2	1	—	Damage growth rate	
					W	1	—	1		
Test No. 12	65C17774	Root lug tests 	508 x 76 x 38 (20 x 3 x 1.5)	-1 Tension	D	2	2	—	Failure load and mode of failure	Static strength tension and compression
					W	2	—	2		
				-2 Compression	D	2	2	2		
					W	—	—	2		
				-1	D	3T, 3C	3T, 3C	—	Failure mode and cyclic life damage growth rates	Spectrum fatigue loads ①; T—tension spectrum; C—compression spectrum
					W	—	—	3T, 3C		

Note: Wet condition denotes 1.1% moisture content achieved by conditioning in a 100% relative humidity chamber at 60°C

① Two of each set will be subjected to two life spectrum fatigue tests followed by static test to failure, and the remaining specimen of each set will be subjected to four life spectrum fatigue tests.

② Lower surface

③ Upper surface

Figure 3-4. Stabilizer Subcomponent Test Plan (Cont)

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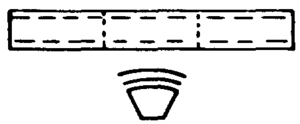
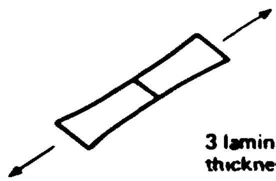
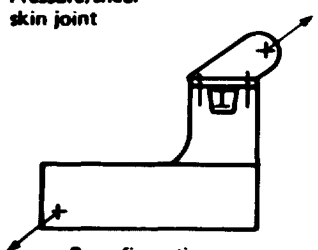
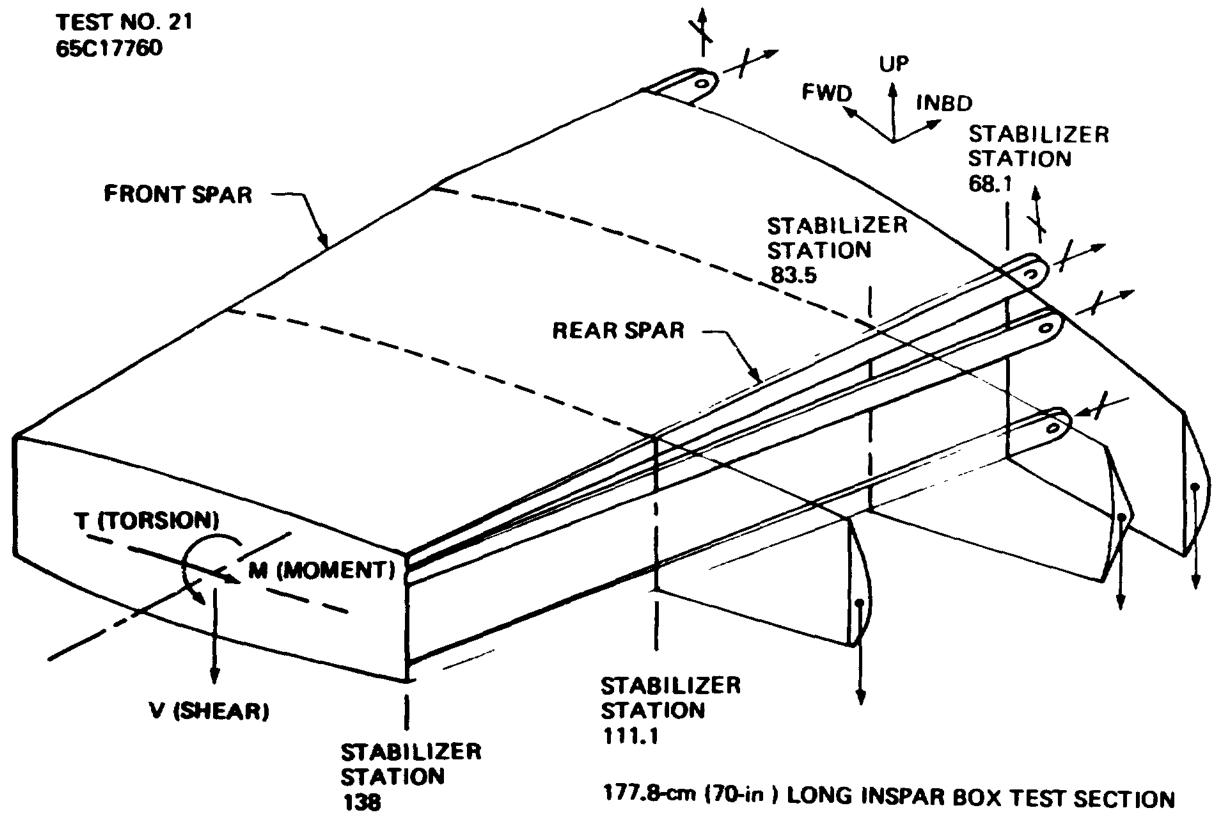
Test No.	Drawing No.	Specimen	Size mm (in.)	Config-uration	Condition Wet (W) Dry (D)	Test temperature and number of tests			Data	Load condition
						Room temp-earture	$-54^{\circ}C$	$+82^{\circ}C$		
Test No. 20	65C17792	Sonic test box 2 skin panels 	762 x 762 (30 x 30)	-1	D	1	—	—	Time to failure	Sonic fatigue
				-2	D	1	—	—		
Test No. 22	65C17980	Discontinuous laminate  3 laminate thickness ratios	305 x 51 (12 x 2)	-1	D	3	3	—	Delam-ination strain levels	Static tension
				-2	D	3	3	—		
				-3	D	3	3	—		
Test No. 24	65C17981	Pressure/shear skin joint  2 configurations	305 x 305 (12 x 12)	-1	D	3	3	—	Joint strength	Static tension
				-2	D	3	3	—		

Figure 3-4. Stabilizer Subcomponent Test Plan (Concl)



TEST SEQUENCE

1. Static test to design-limit-load conditions
2. Spectrum fatigue test (1¼ lifetimes)
3. Static test to design-ultimate-load conditions
4. Failsafe load test (three damaged areas)
5. Destruction test (critical condition)

Figure 3-5. Design Development Test Stub Box

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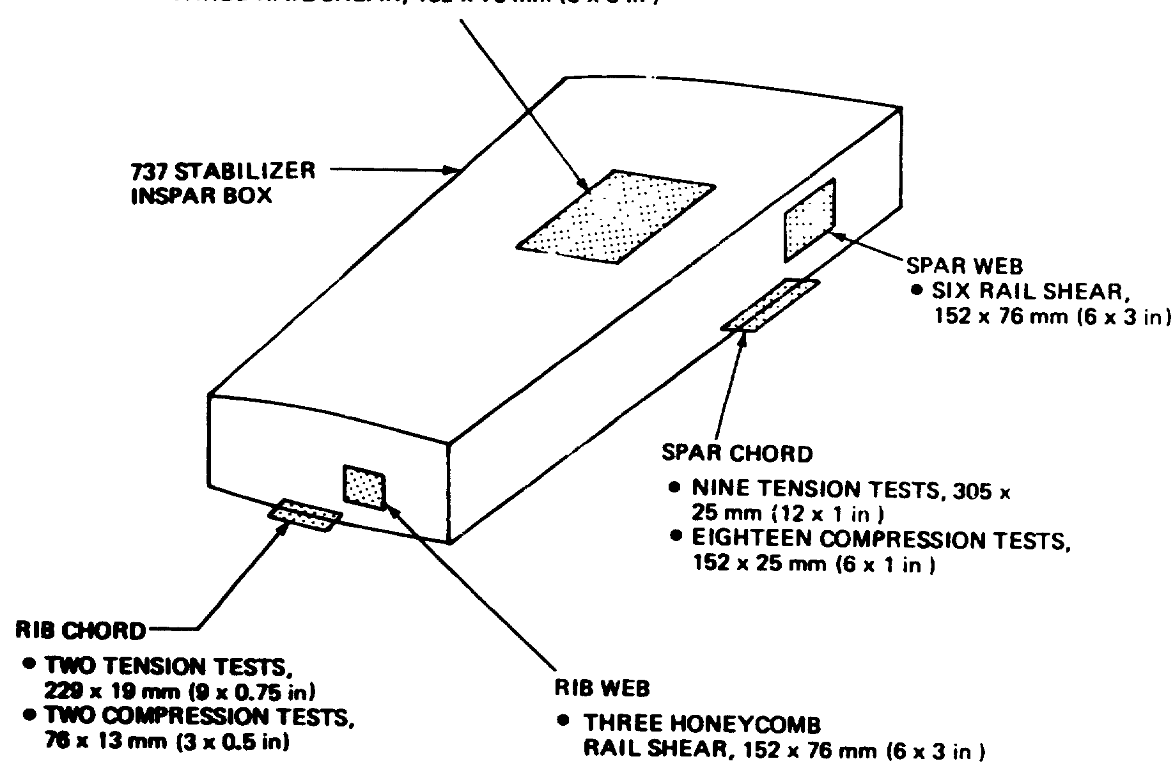

NOTE: Specimens taken from routine manufactured parts.
All tests conducted at room temperature.

Figure 3-6. Testing of Production Verification Hardware—Test No. 25

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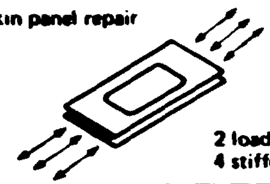
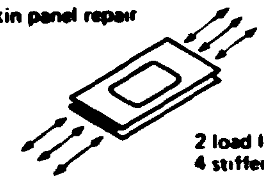
Test No.	Drawing No.	Specimen	Size, mm (in.)	Configuration	Condition Wet (W) Dry (D)	Test temperature and number of tests: Room temperature	-54°C	+82°C	Data	Load condition
Test No. 15	65C17787	Skin panel repair  2 load levels 4 stiffeners	1397 x 345 (55 x 13.6)	-1	D	2	2	–	Failure load and mode of failure	Static strength
					W	2	–	2		
				-2	D	2	2	–		
Test No. 15	65C17787	Skin panel repair  2 load levels 4 stiffeners	762 x 345 (30 x 13.6)	-1	D	3	3	–	Failure load and mode of failure [2]	Spectrum fatigue load [1]
					W	3	–	3		
				-2	D	3	–	3		

Note: Wet condition denotes 1.1% moisture content achieved by conditioning in a 100% relative humidity chamber at 60°C

[1] Two of each set will be subjected to two life spectrum fatigue tests followed by static test to failure, and the remaining specimen of each set will be subjected to four life spectrum fatigue tests.

[2] Damage growth rates measured.

Figure 3-7. Maintenance and Repair Test Plan

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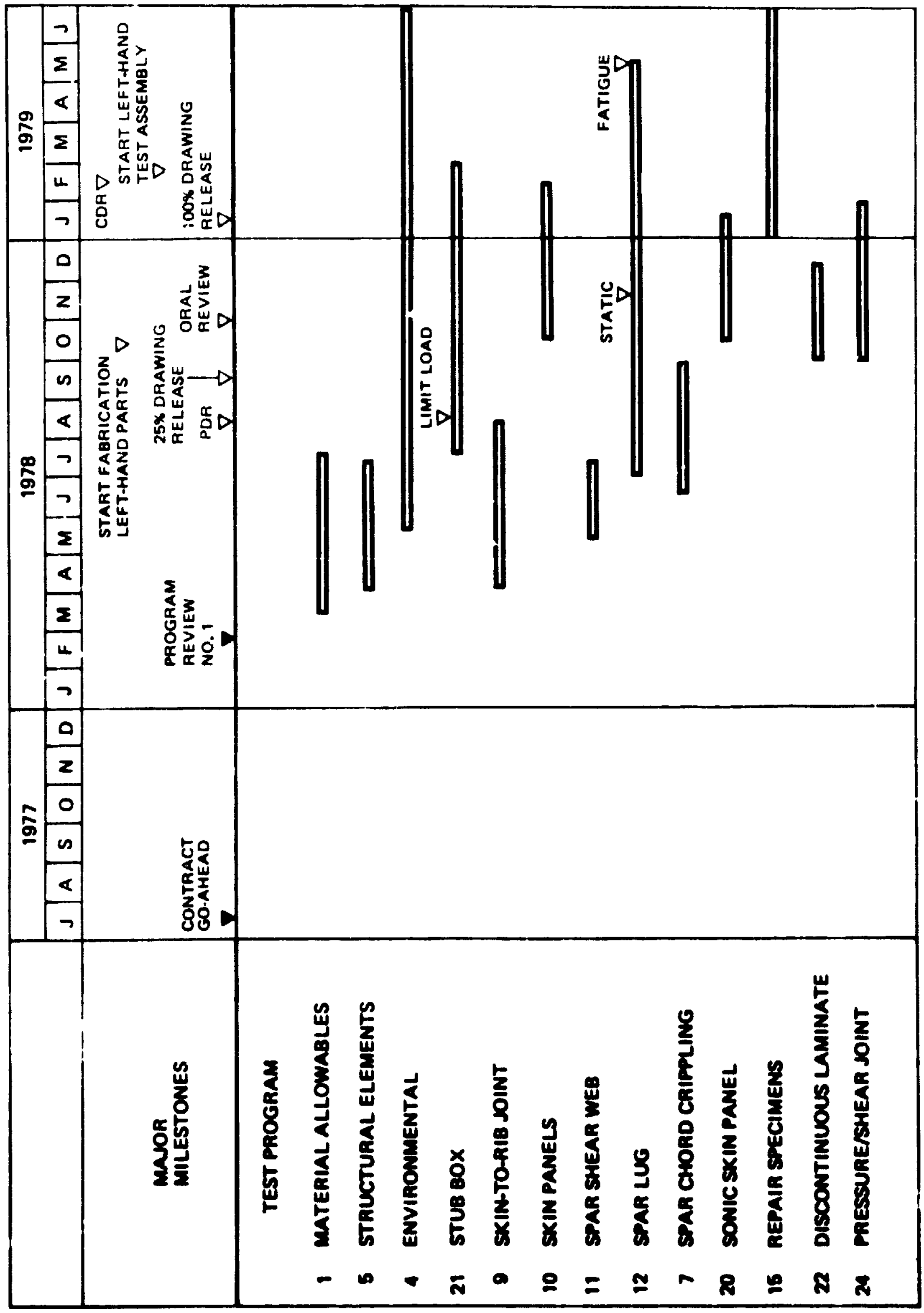

Figure 3-8. 737 Advanced Composites Stabilizer Ancillary Test Plan Schedule

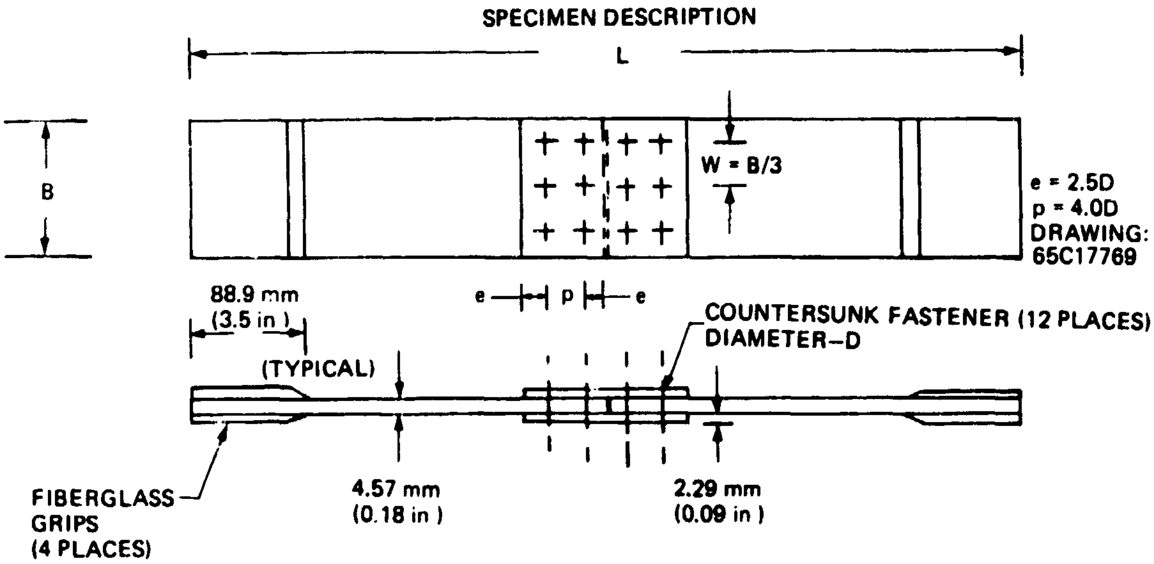

Figure 3-9. 50% Load Transfer Joint

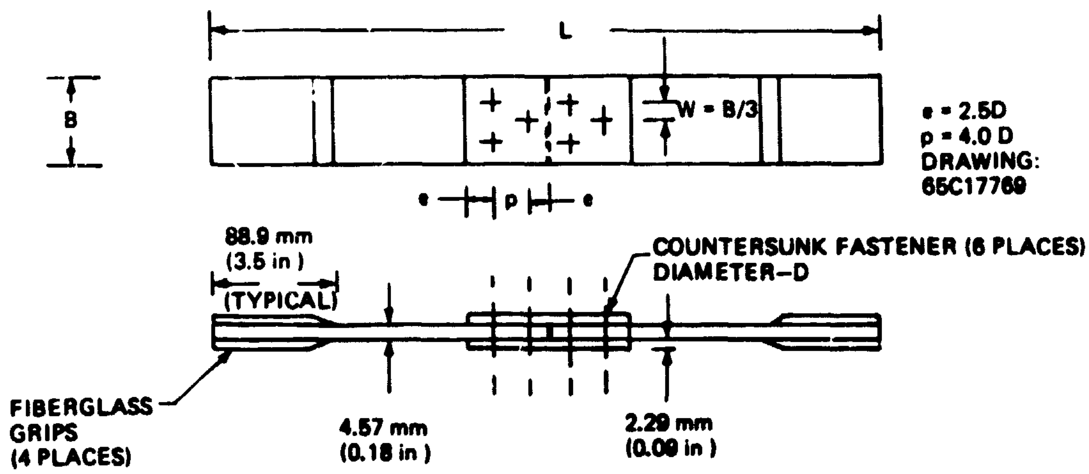

Figure 3-10. 100% Load Transfer Joint

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Table 3-1. *50% Load Transfer Joint Test Results—Test No. 5*

Drawing No. 65C17769 Assembly No.	Fastener diameter, mm (in)	Specimen geometry L, mm (in)	Specimen geometry B, mm (in)	W/D	Fabric layup [1]	Failure load, [2] N	Failure load, (lb)	End load at failure, kN/m	(kips/in)
-1	4.76 (3/16)	549.3 (21.63)	71.4 (2.81)	5	[(0/90) (±45)]	67,254 66,542 66,809	(15,120) (14,960) (15,020)	942 932 936	(5.4) (5.3) (5.3)
-2		663.6 (26.13)	100.1 (3.94)	7		85,224 80,776 79,174	(19,160) (18,160) (17,800)	851 807 791	(4.9) (4.6) (4.5)
-3	6.35 (1/4)	673.1 (26.50)	95.3 (3.75)	5		84,734 81,398 82,466	(19,050) (18,300) (18,540)	889 854 865	(5.1) (4.9) (4.9)
-4		825.5 (32.50)	133.4 (5.25)	7		96,077 105,418 102,304	(21,600) (23,700) (23,000)	720 790 767	(4.1) (4.5) (4.4)

[1] Material: Narmco 5208 7-mil fabric
[2] Failures in countersunk splice plate

- Environmental condition—dry
- Static tension
- Test temperature, 21°C (70°F)

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Table 3-2. 100% Load Transfer Joint Test Results—Test No. 5

Drawing No. 65C17769 Assembly No.	Fastener diameter, mm (in)	Specimen geometry L, mm (in)	Specimen geometry B, mm (in)	W/D	Fabric layup [1]	Failure load, [2] N (lb)	End load at failure, kN/m (kips/in)
-5	4.76 (3/16)	434.9 (17.13)	42.67 (1.68)	3	[(0/90)(±45)$_2$]	35,495 (7,980) 34,249 (7,700) 34,072 (7,660)	832 (4.8) 803 (4.6) 799 (4.6)
-6	↓	549.3 (21.63)	71.37 (2.81)	5	↓	41,455 (9,340) 38,786 (8,720) 41,544 (9,340)	581 (3.3) 543 (3.1) 582 (3.3)
-7	6.35 (1/4)	520.7 (20.50)	57.15 (2.25)	3	↓	40,210 (9,040) 40,566 (9,120) 44,035 (9,900)	704 (4.0) 710 (4.1) 771 (4.4)
-8	↓	673.1 (26.50)	95.25 (3.75)	5	↓	56,134 (12,620) 52,931 (11,900) 52,486 (11,800)	589 (3.4) 555 (3.2) 551 (3.1)

[1] Material: Narmco 5208 7-mil fabric
[2] Failures in countersunk splice plate

- Environmental condition—dry
- Static tension
- Test temperature, 21°C (70°F)

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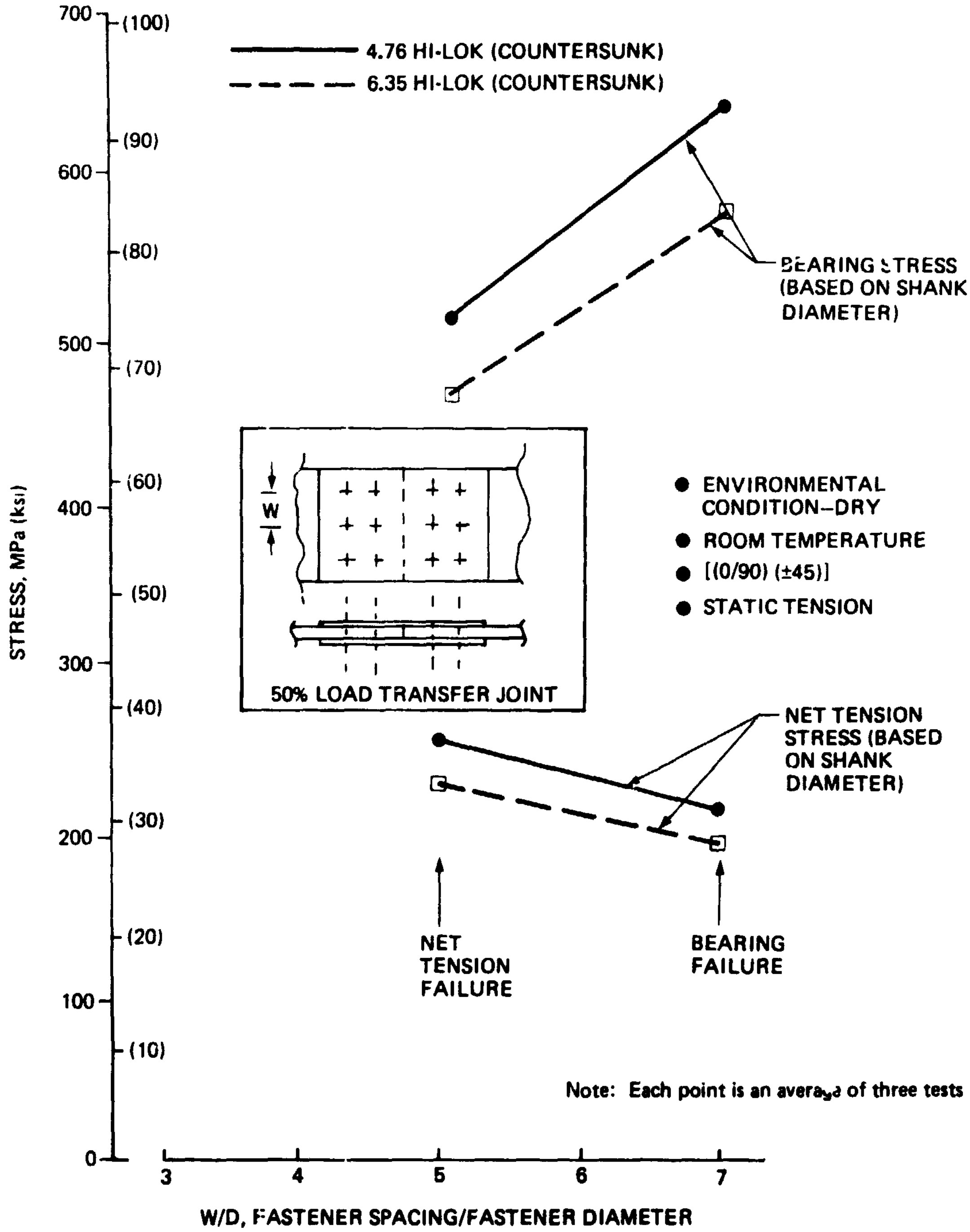

Figure 3-11. Net Tension and Bearing Stresses at Failure

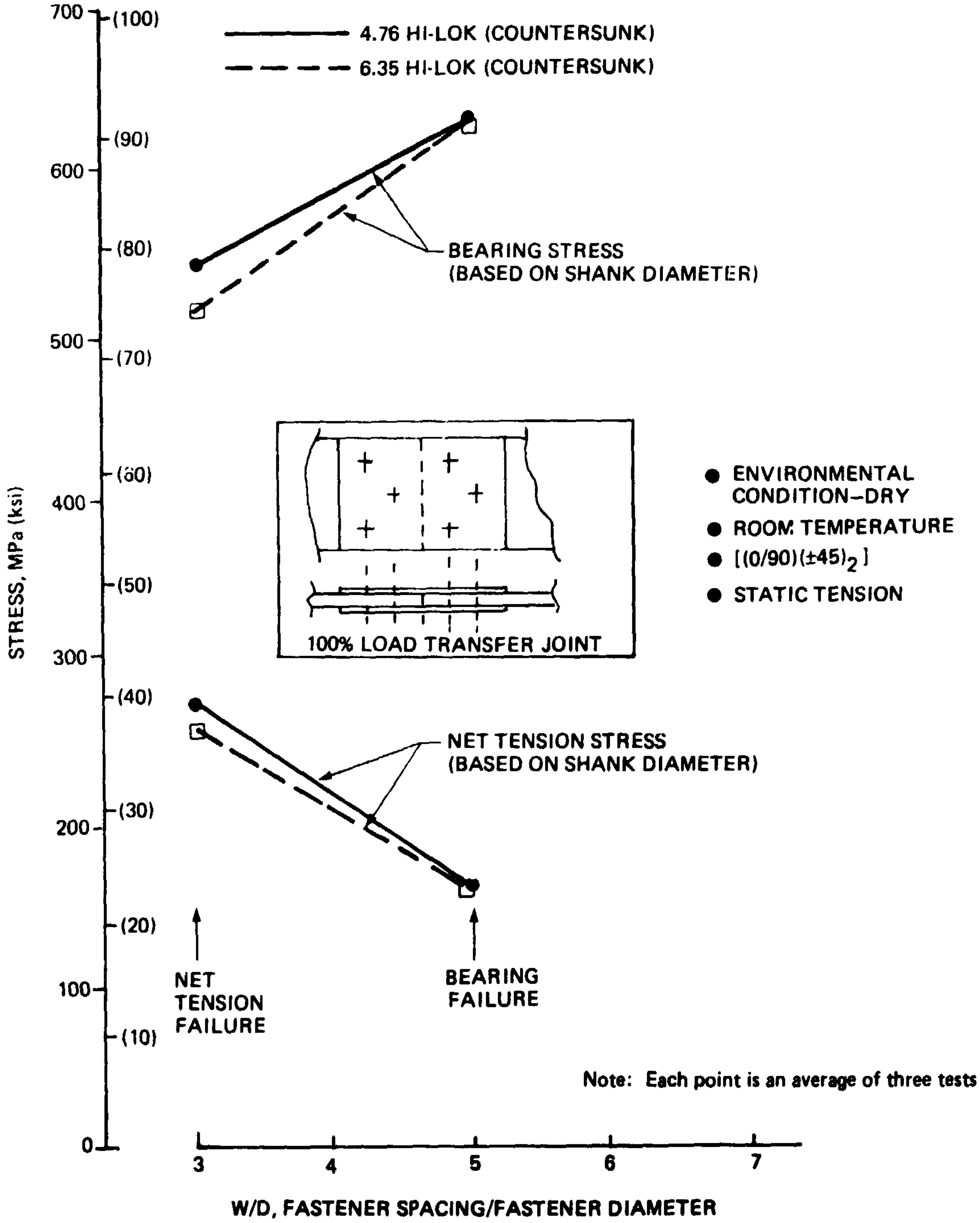

Figure 3-12. Net Tension and Bearing Stresses at Failure

Results for the Test No. 1 bolted joint tests presented in Reference 6, and present results from Test No. 5, have been evaluated by comparing the load/mm (load/in) capability for the single- and multiple-width fastener joints. The test results from Reference 6 are presented in Tables 3-3 and 3-4, and these results are compared to the Test No. 5 results in Figures 3-13 and 3-14. The comparison of both the 100% and 50% load transfer joints indicates that the single-width fastener joint is a reasonable representation of a multiple-width fastener joint, for values of W/D where bearing failure is the primary mode of failure.

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Table 3-3. 50% Load Transfer Joint Test Results—Test No. 1

Drawing No. 65C17768 Assembly No.	Fastener diameter, mm (in.)	Specimen geometry L, mm (in.)	Specimen geometry W, mm (in.)	W/D	Fabric layup [1]	Failure load [2], N	Failure load [2], (lb)	End load at failure, kN/m	End load at failure, (kips/in.)
-11	4.76 (3/16)	359.16 (14.14)	23.88 (0.94)	5	[(0/90)(±45)]	20,372 20,950 21,306	(4,580) (4,710) (4,790)	853 877 892	(4.9) (5.0) (5.1)
-12		396.75 (15.62)	33.27 (1.31)	7		28,556 27,489 25,887	(6,420) (6,180) (5,820)	858 826 778	(4.9) (4.7) (4.4)
-13	6.35 (1/4)	419.10 (16.50)	31.75 (1.25)	5		24,642 26,332 26,154	(5,540) (5,920) (5,880)	776 829 824	(4.4) (4.7) (4.7)
-14		469.90 (18.50)	44.45 (1.75)	7		32,915 32,115 32,559	(7,400) (7,220) (7,320)	740 722 732	(4.2) (4.1) (4.2)

[1] **Material: Narmco 5208 7-mil fabric**
[2] **Failures in countersunk splice plate**

- Environmental condition—dry
- Static tension
- Test temperature, 21°C (70°F)

Table 3-4. 100% Load Transfer Joint Test Results—Test No. 1

Drawing No. 65C17768 Assembly No.	Fastener diameter, mm	(in)	Specimen geometry: L, mm	(in)	W, mm	(in)	W/D	Fabric layup [1]	Failure load [2], N	(lb)	End load at failure, kN/m	(kips/in)
-7	4.76	(3/16)	282.45	(11.12)	14.22	(0.56)	3	[(0/90)(±45)$_2$]	9,452 9,274 9,252	(2,125) (2,085) (2,080)	665 652 651	(3.8) (3.7) (3,7)
-8	↓	↓	321.06	(12.64)	23.88	(0.94)	5	↓	12,632 12,832 12,944	(2,840) (2,885) (2,910)	529 537 542	(3,0) (3.1) (3.1)
-9	6.35	(1/4)	317.50	(12.50)	19.05	(0.75)	3	↓	13,589 13,722 12,388	(3,055) (3,085) (2,785)	713 720 650	(4.1) (4.1) (3.7)
-10	↓	↓	368.30	(14.50)	31.75	(1.25)	5	↓	17,925 19,060 18,771	(4,030) (4,285) (4,220)	565 600 591	(3.2) (3.4) (3.4)

[1] Material: Narmco 5208 7-mil fabric
[2] Failures in countersunk splice plate

- Environmental condition—dry
- Static tension
- Test temperature, 21°C (70°F)

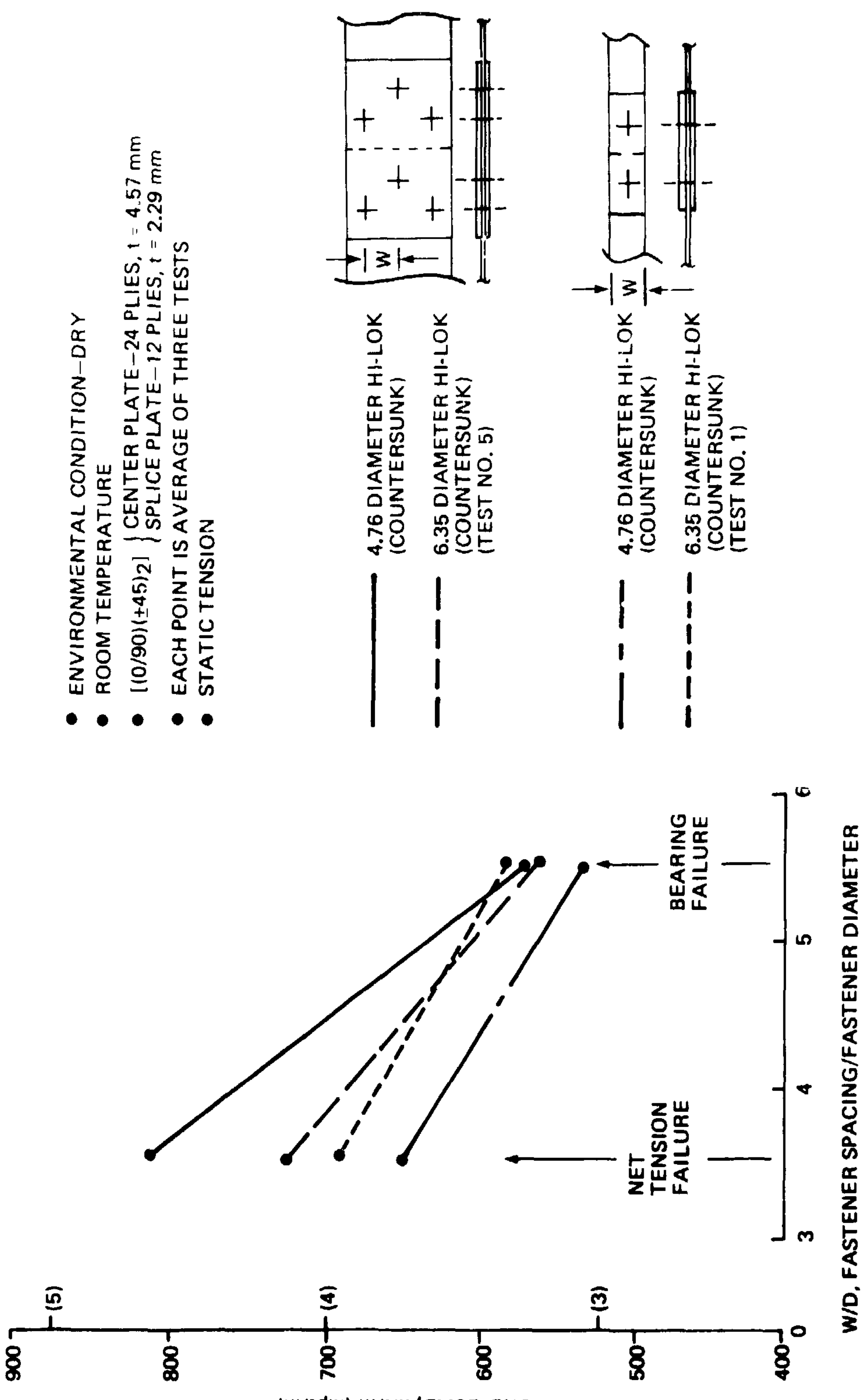

Figure 3-13. 100% Load Transfer Joint

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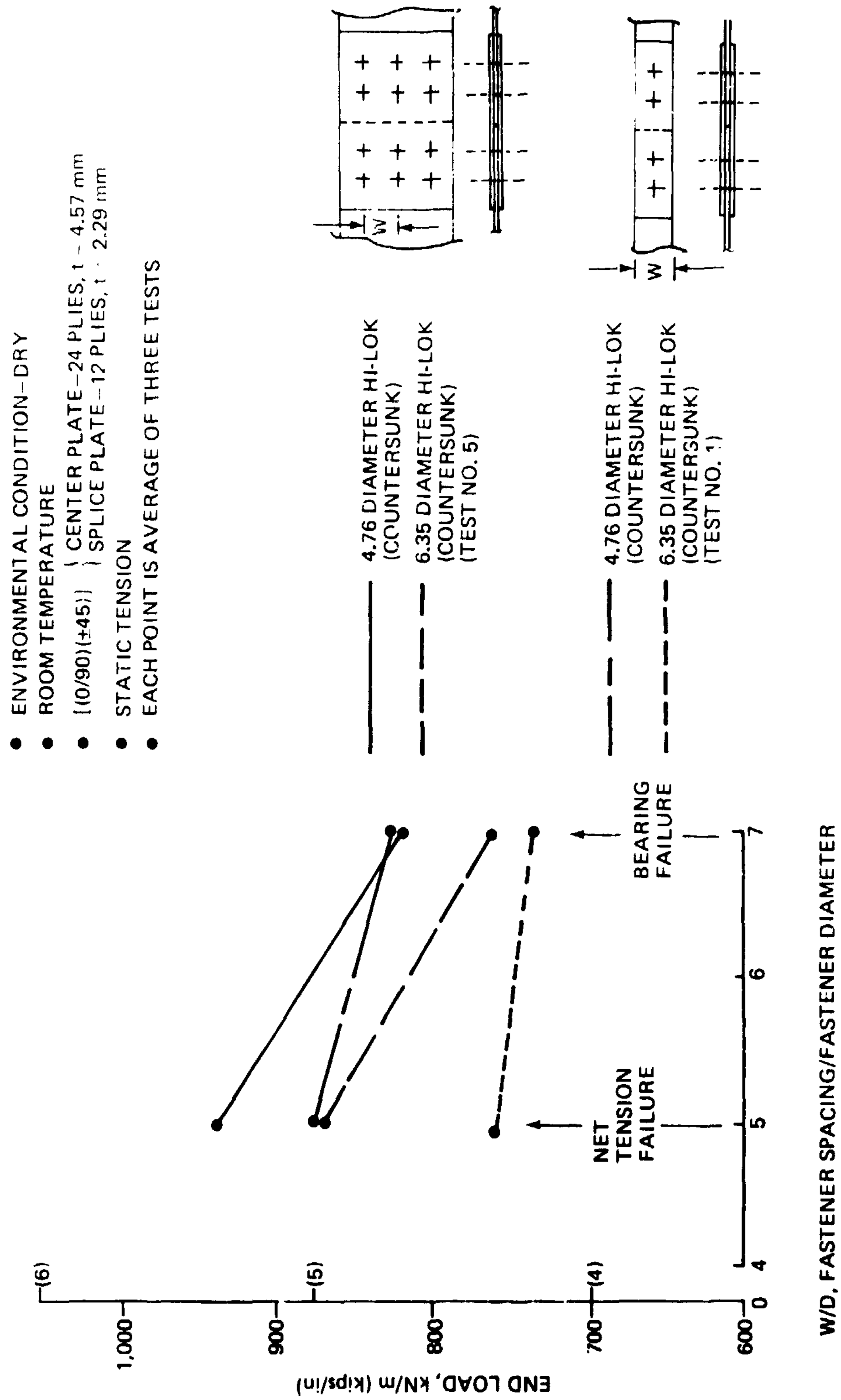

Figure 3-14. 50% Load Transfer Joint

SECTION 4.0

OPERATIONS DEVELOPMENT

This section discusses results of the manufacturing producibility studies, ancillary test component fabrication and manufacturing, quality assurance development efforts, and verification hardware.

4.1 PRODUCIBILITY STUDIES

The producibility study on the rear spar/lug interface test section has been completed.

The previous quarterly reports described the detail fabrication and Verifilm process. This report includes the spar detail bonding, machining, and drilling, and attachment of the titanium lugs.

The bonding was accomplished in two operations as detailed in Figure 4-1. During the first-stage bond, a layer of Verifilm was used between the caul plates and web flanges, to minimize friction, and ensure good mating of the bonding surfaces. The graphite/epoxy tape filler, applied to the radii of the webs, and the filler/cap details were cocured in the final stage. Envelope bagging was used for the final stage in order to avoid the need for a recessed tool to accommodate the filler previously bonded to the cap.

After bonding, the flanges of the spar were checked for flatness. It was determined that there was a downward bow up to 0.060 cm (0.024 in) in some areas of the flanges. This bow generally extended from the flange tip to about 1.52-1.78 cm (0.06-0.70 in) inward. The bowing occurred as a result

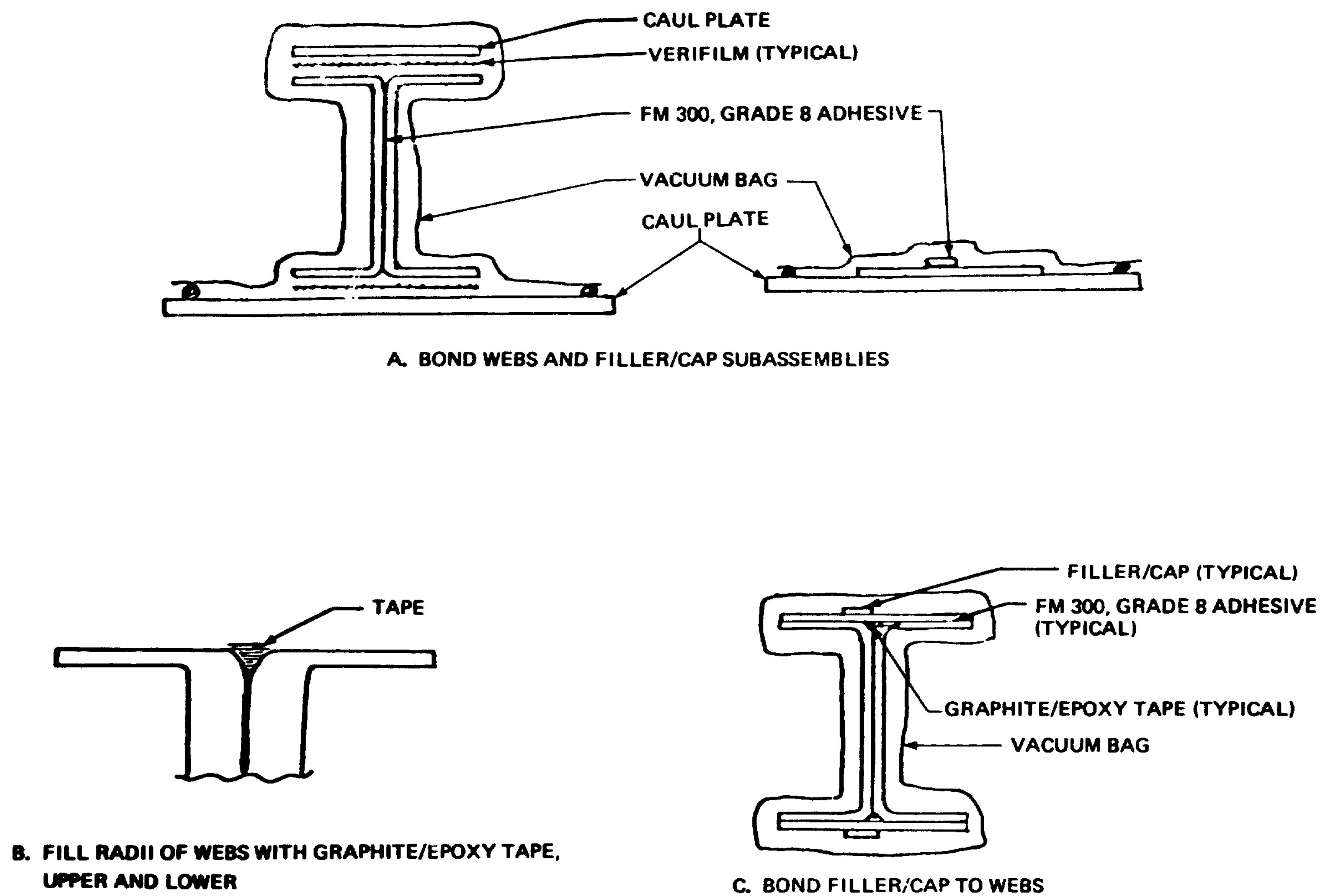

Figure 4-1. Spar/Lug Feasibility Hardware, Showing the Bonding Operation for the Details and Filler/Cap

of bonding the two webs together, because the web bond surfaces were concave in the chordwise direction prior to bonding. This bowing will cause no problem in the fabrication of the verification or production hardware.

Dimensional measurements indicated that thickness can be controlled to drawing tolerances. The spar measured 2.941 cm (1.158 in) at the thickest point of the graphite/epoxy lug area. The feasibility spar was made in a female aluminum tool with fixed legs. The verification and production hardware tooling will utilize a movable leg concept, to prevent the tool legs from compressing against the webs during cure-cycle cooling. This tooling concept is expected to reduce warpage in the chordwise direction.

Machining of the graphite/epoxy lug area (Figure 4-2) was accomplished on a profile mill using a diamond cutter. The titanium/graphite/epoxy lug fastener holes were piloted using a carbide tip drill. The titanium lugs were then bonded to the graphite/epoxy using bolts through the pilot holes for pressure. Figures 4-3 and 4-4 illustrate the polysulfide adhesive being applied.

The final fastener hole size was obtained using a carbide tip drill and a carbide reamer. A boring hole and a carbide tool were used for the bushing holes. Figures 4-5 through 4-7 show the boring of the bushing holes.

4.2 ANCILLARY TEST COMPONENT FABRICATION

The ancillary test plan includes allowables and environmental, concept verification, and repair. The following describes the fabrication status of each effort as of June 29, 1978.

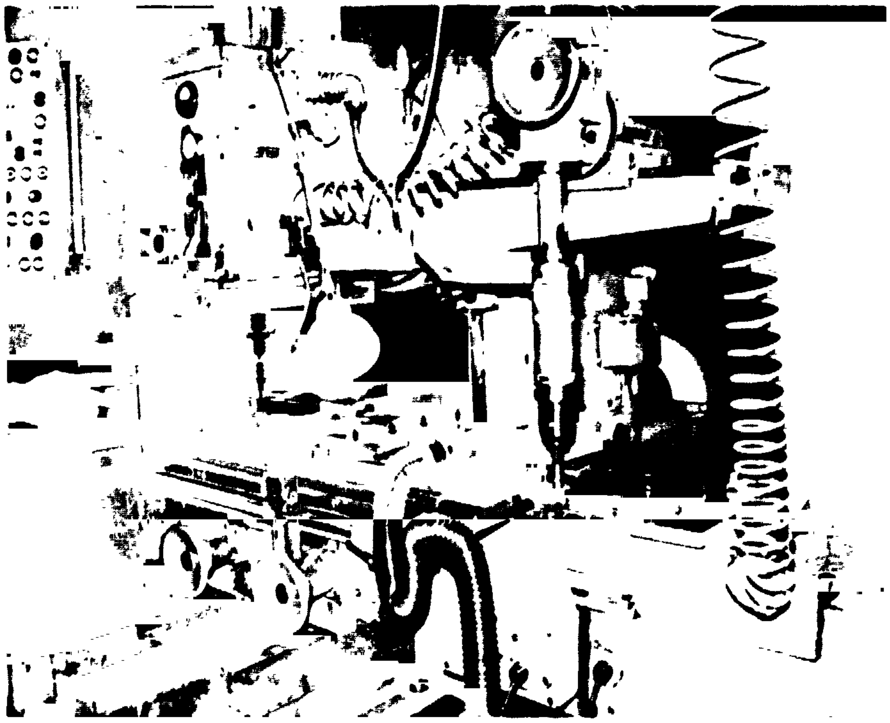

Figure 4-2. Spar/Lug Feasibility Hardware, Showing Machining of Graphite/Epoxy Lugs Using a Profile Mill

Figure 4-3. Spar/Lug Feasibility Hardware, Showing Polysulfide Adhesive Being Applied For Bonding Titanium Lug

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Figure 4-4. Spar/Lug Feasibility Hardware, Showing Titanium Lug Being Bonded

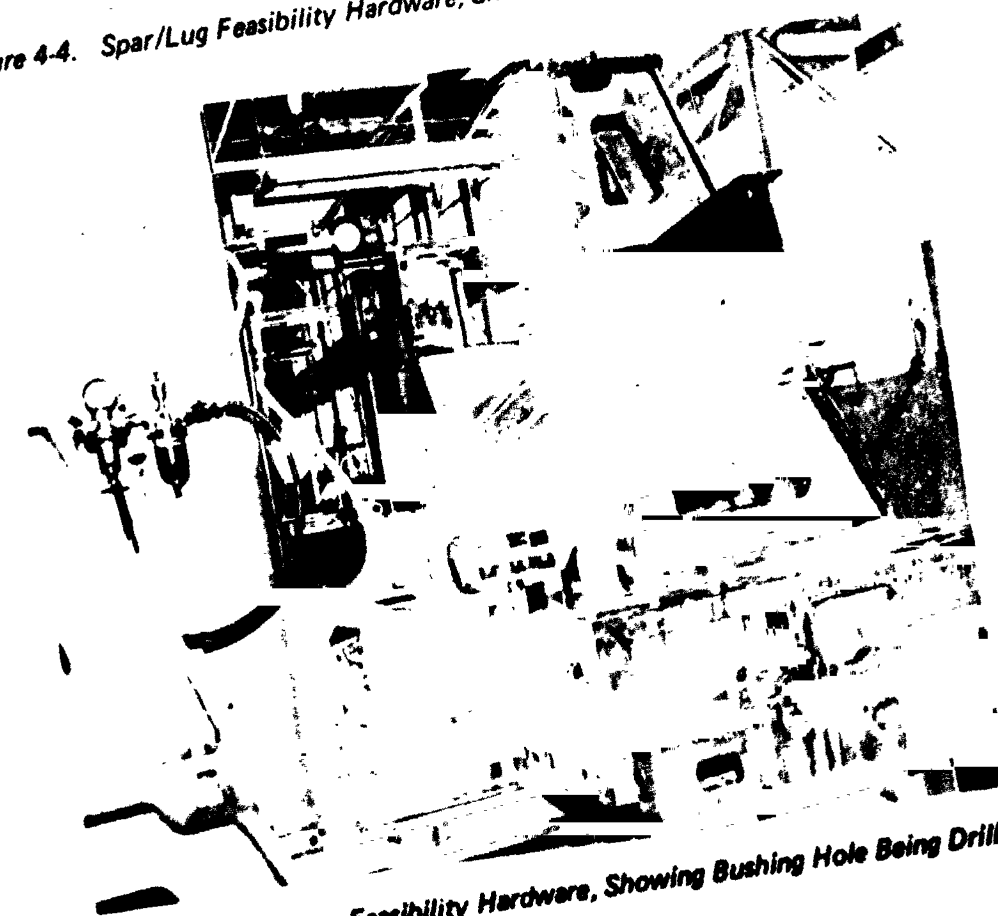

Figure 4-5. Spar/Lug Feasibility Hardware, Showing Bushing Hole Being Drilled

Figure 4-6. Spar/Lug Feasibility Hardware, Showing Bushing Hole Being Drilled

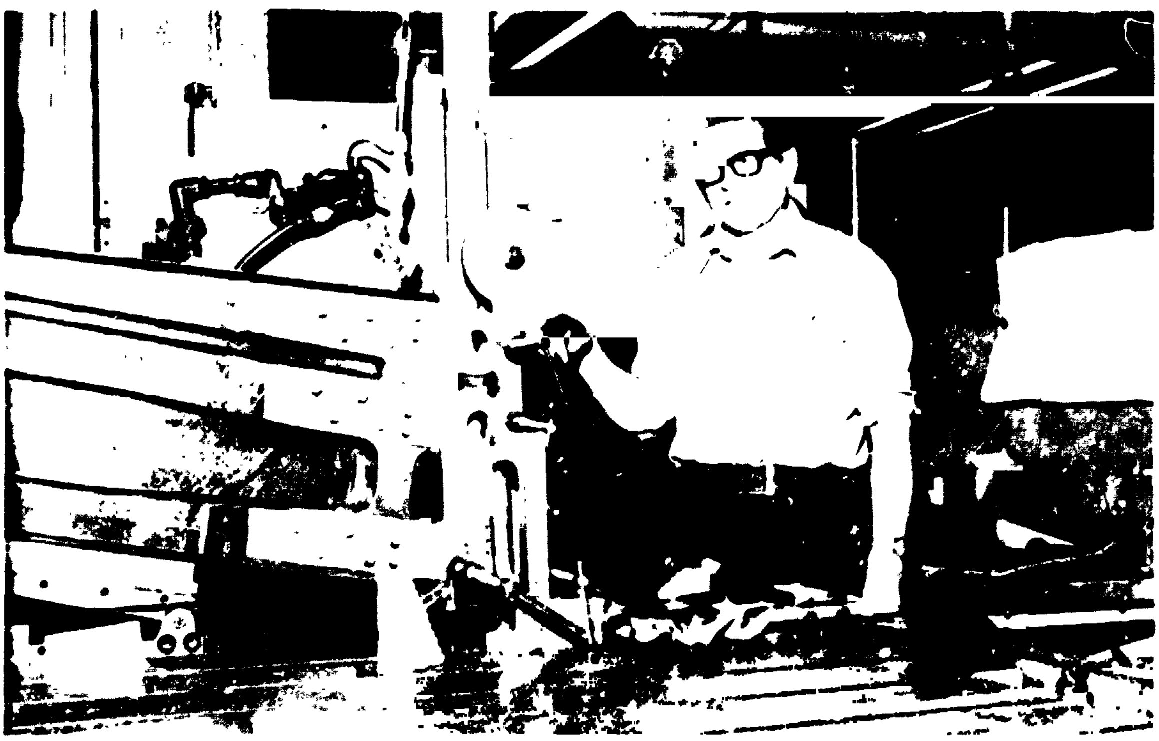

Figure 4-7. Spar/Lug Feasibility Hardware, Showing Finishing Cut On Bushing Hole

4.2.1 Allowables and Environmental

This part of the ancillary test program includes material allowables (Test No. 1), mechanical joints (Test No. 5), and environmental specimens (Test No. 4).

The detail fabrication and assembly for the allowables (Test No. 1), mechanical joints (Test No. 5) and environmental specimens (Test No. 4) are complete.

Specimens that were rejected because of tolerance problems related to fiberglass grip tab bonding, prior to envelope bagging, were reworked rather than remade. Figure 4-8 shows typical reworked specimens.

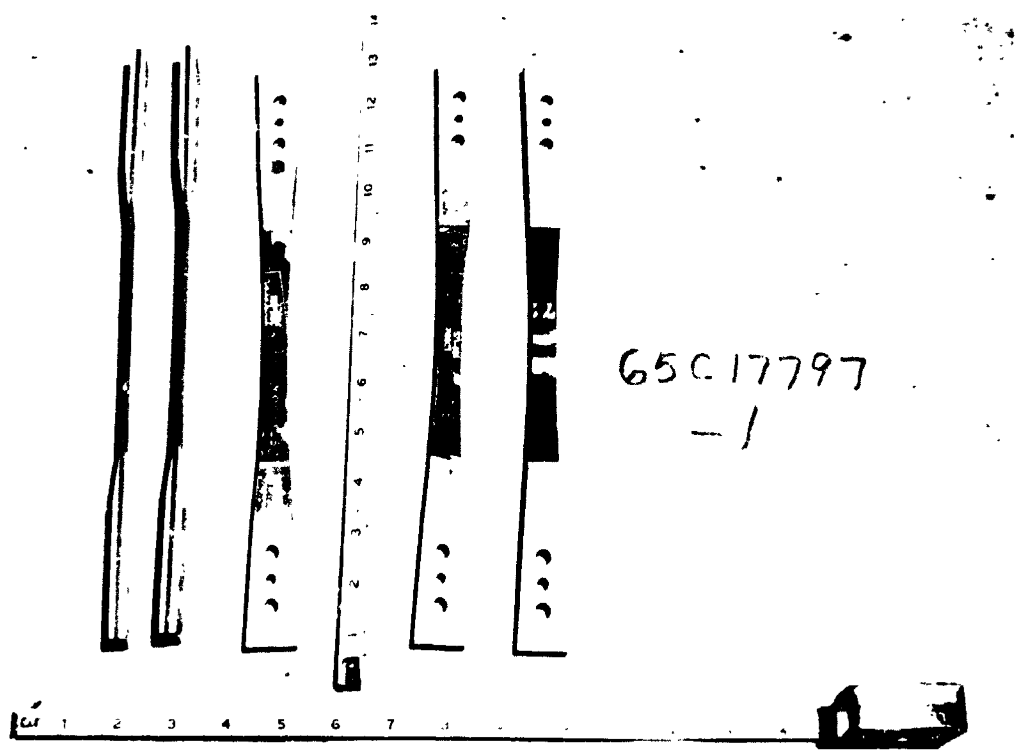

Figure 4-8. Ancillary Test Allowables, Showing Typical Reworked Specimens

4.2.2 Concept Verification

This part of the ancillary test program includes spar chord crippling (Test No. 7), skin-to-rib joints (Test No. 9), skin panel (Test No. 10), spar shear web (Test No. 11), spar lug (Test No. 12), sonic test box (Test No. 20), stub box (Test No. 21), discontinuous laminate critical strain (Test No. 22), skin-panel-to-rib joint (Test No. 24), production-verification (Test No. 25), and manufacturing feasibility spar test coupons (Test No. 26). The following describes the fabrication and assembly status:

- Spar chord crippling (Test No. 7)
  Detail fabrication and assembly complete, (Figures 4-9 and 4-10).

- Skin-to-rib joints (Test No. 9)
  Detail fabrication and assembly complete.

- Skin panel (Test No. 10)
  Tool fabrication 30% complete.

- Spar shear web (Test No. 11)
  Detail fabrication and assembly complete.

- Spar lug (Test No. 12)
  Detail fabrication complete, Assembly 50% complete (Figures 4-11 through 4-16).

- Sonic test box (Test No. 20)
  In Tool and Production Planning.

- Stub box (Test No. 21)
  Detail fabrication complete. Assembly 50% complete.

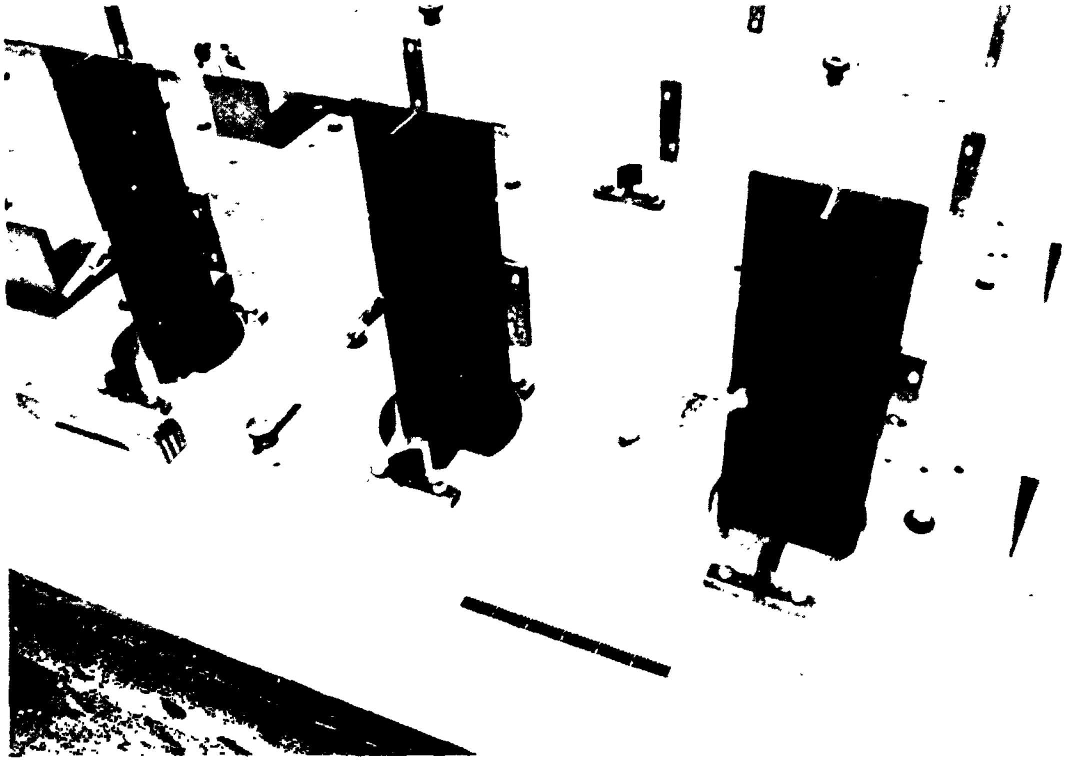

Figure 4-9. Spar Chord Crippling (Test No. 7), Showing Specimen Ready for End Potting

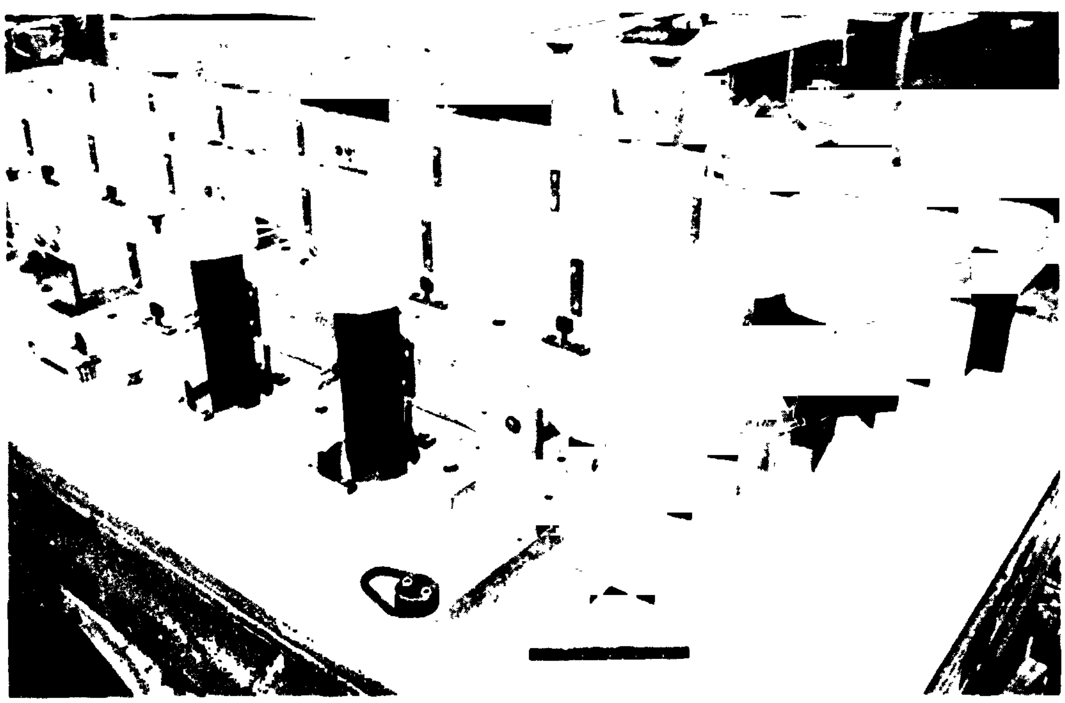

Figure 4-10. Spar Chord Crippling (Test No. 7), Showing Completed Specimens

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Figure 4-11. Spar Lug (Test No. 12), Showing Specimens Ready for Cure

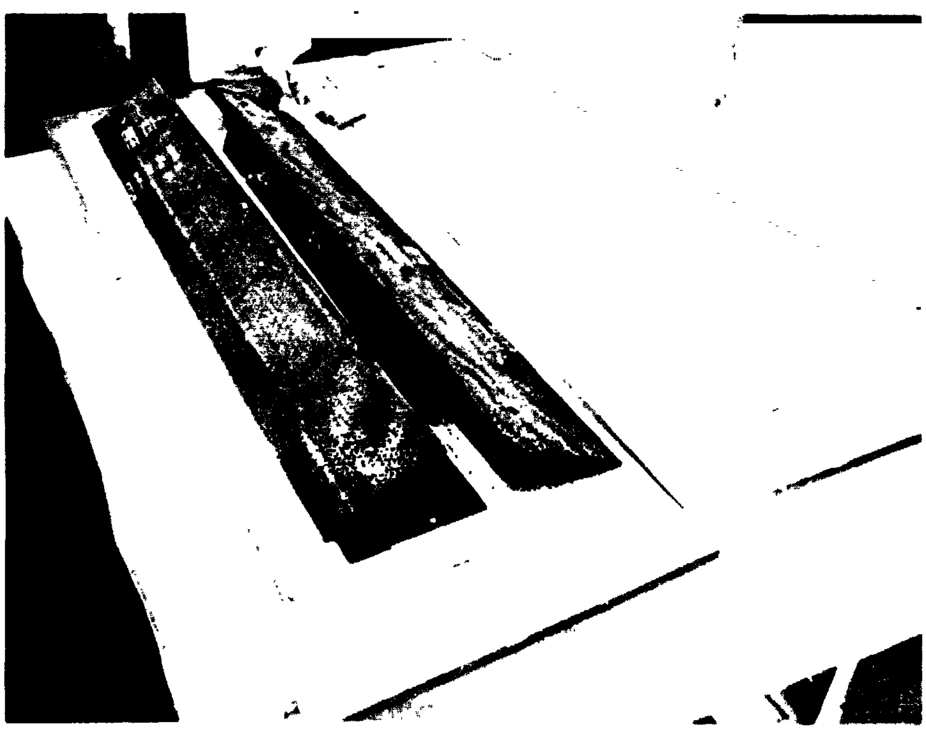

Figure 4-12. Spar Lug (Test No. 12), Showing Peel Ply Being Removed from Completed Detail Halves

Figure 4-13. Spar Lug (Test No. 12), Showing Completed Detail Halves Bagged and Ready for Bonding

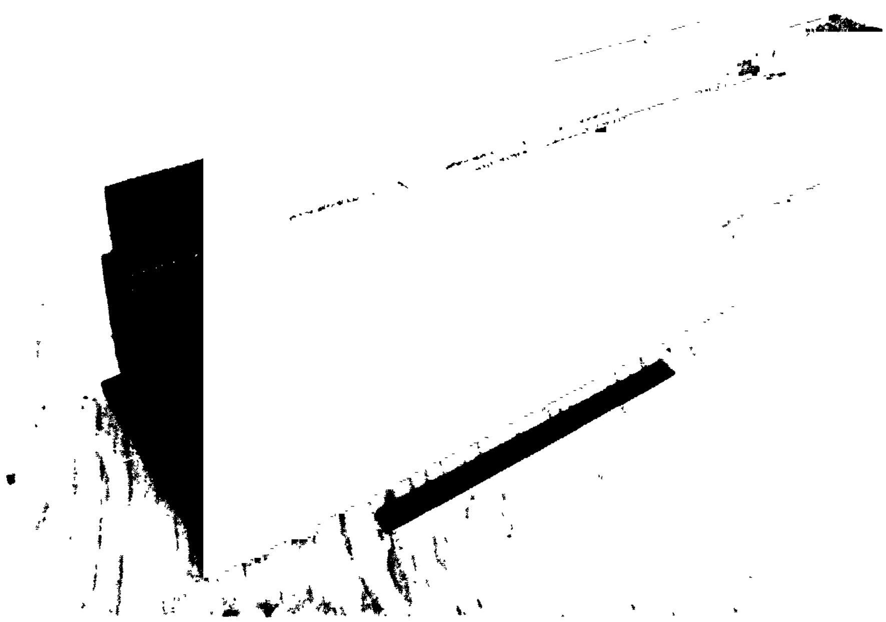

Figure 4-14. Spar Lug (Test No. 12), Showing Trimmed Compression Specimen

Figure 4-15. Spar Lug (Test No. 12), Showing Trimmed Tension Specimen

Figure 4-16. Spar Lug (Test No. 12), Showing Drilling of Fastener Holes

- Discontinuous laminate critical strain (Test No. 22)
  In Tool and Production Planning.

- Skin-panel-to-rib joint (Test No. 24)
  In Tool and Production Planning.

- Production-verification (Test No. 25)
  In Tool and Production Planning.

- Manufacturing feasibility spar test coupons (Test No. 26)
  Planning complete. Test coupons and NDI rear spar standards being cut.

4.3 QUALITY ASSURANCE DEVELOPMENT

This section discusses the evaluation of the preliminary NDI standards, and the fabrication of the production NDI standards.

The preliminary NDI standards have been completed. The NDI techniques evaluated were as follows:

- X-ray
- Through-Transmission Ultrasonic
- Sondicator
- Fokker Bond Tester

Details of the preliminary investigation will be provided in a later report containing the production standards. In summary, all preliminary standard defects were detected by one or more NDI techniques. It is concluded that Through-Transmission Ultrasonic technique can be used to detect 0.64 x 0.64 cm (0.25 x 0.25 in) defects in arts. X-ray inspection is recommended for radii areas. Initial results indicate that in-service inspection can be conducted by the Sondicator and/or the Fokker Bond Tester for laminated structure, and by the Sondicator for honeycomb structure.

The production NDI standards are now being fabricated as Task III of Test No. 25, production-verification. The production NDI standards will include a section of the upper and lower skin panel, a section of two ribs (honeycomb and laminate), and a section of the front spar. Preliminary tests indicated that a section of the feasibility rear spar can be used for the rear spar production standard.

4.4 VERIFICATION HARDWARE

The stub box (Test No. 21) is being used for the verification hardware. The stub box is a full-scale root section of the advanced composites 737 horizontal stabilizer. It consists of the structural box from the side-of-body outboard to Station 152.45, including the trailing-edge structure and closure rib.

During fabrication of the graphite/epoxy details for the verification hardware (stub box, Test No. 21) the following problems were encountered and resolved:

- Rear Spar Fabrication

  Both details for the rear spar were rejected and scrapped because of bagging problems that caused the bag to fail during cure, in addition to excessive detail resin bleed-out.

  An investigation of the problem indicated the bag was bridged, while the excessive resin bleed-out was caused by the numerous pleats at the end of the part.

  To ensure that these problems do not occur again, the following action was initiated:

  - The area supervisor and manufacturing shop support personnel will check all bags on complex parts to ensure they are not bridged.

- Vacuum bag sealant will be used around the end of the part as a dam to eliminate the resin bleed-out. This new processing procedure has been added to the Boeing Process Specification BAC 5562.

- Rib Fabrication

The verification ribs have been rejected because of build-ups in the corner areas. The drawing allows only overlap splices in the corner areas, and Manufacturing concluded they cannot guarantee a rib without this build-up.

The verification rib build-ups will be removed by sanding, to eliminate any interference problems. However, the production ribs will require a design change. Engineering will revise the production drawings, to remove the corners and extend the joggle areas to obtain additional fastener edge margin lost from the removal of the corner. The build-up areas and the current production design change were shown in Section 2.0 (see Figure 2-7).

- "I" Stiffened Skin Panel Fabrication

Excessive warpage and porosity problems were encountered with the verification "I" stiffened skin panels. Both the upper and lower skin panels warped in excess of 1.40 cm (0.55 in). It took approximately 630 kg (1400 lb) pressure to bring the skin panels back into contour. It was concluded that unbalanced "I" stiffeners are large contributors to the warpage problem. Because warpage has been a major problem with graphite/epoxy, a Boeing-funded program on warpage has been initiated. This program will use a structure similar to the "I" stiffened panel for study purposes.

The upper "I" stiffened skin panel was rejected and scrapped because of excess porosity on the tool surface. A review of the problem indicated that the single fiberglass yarn used per the specification to evacuate the air was not sufficient. The specification has been revised to allow additional fiberglass yarns as the area of the panel increases. In an effort to correlate the effect of porosity, Engineering will cut and test specimens from the scrapped panel.

A second upper skin panel was fabricated, using additional fiberglass yarns between the layup and edge breather, and the porosity problem was eliminated. However, the second upper skin panel did warp 1.40 cm (0.55 in) in the spanwise direction.

Figures 4-17 through 4-26 show fabrication of the rear spar, front spar, and "I" stiffened skin panel.

All details have been fabricated and the sub box is presently being assembled. Figures 4-27 and 4-28 show the start of the stub box (Test No. 21) assembly.

Figure 4-17. Stub Box (Test No. 21) Rear Spar, Showing Incorporation of Precured Insert Into Layup

Figure 4-18. Stub Box (Test No. 21) Front Spar, Showing Completed Details Being Inspected

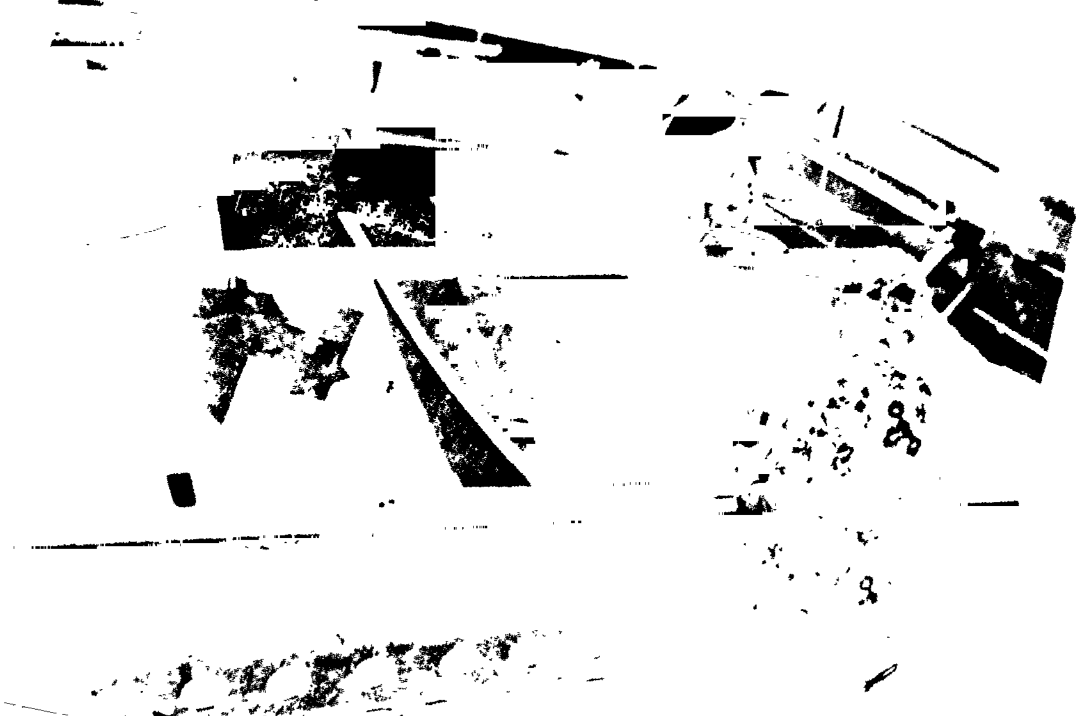

Figure 4-19. Stub Box (Test No. 21) "I" Stiffened Skin Panel, Showing Layup of Skin

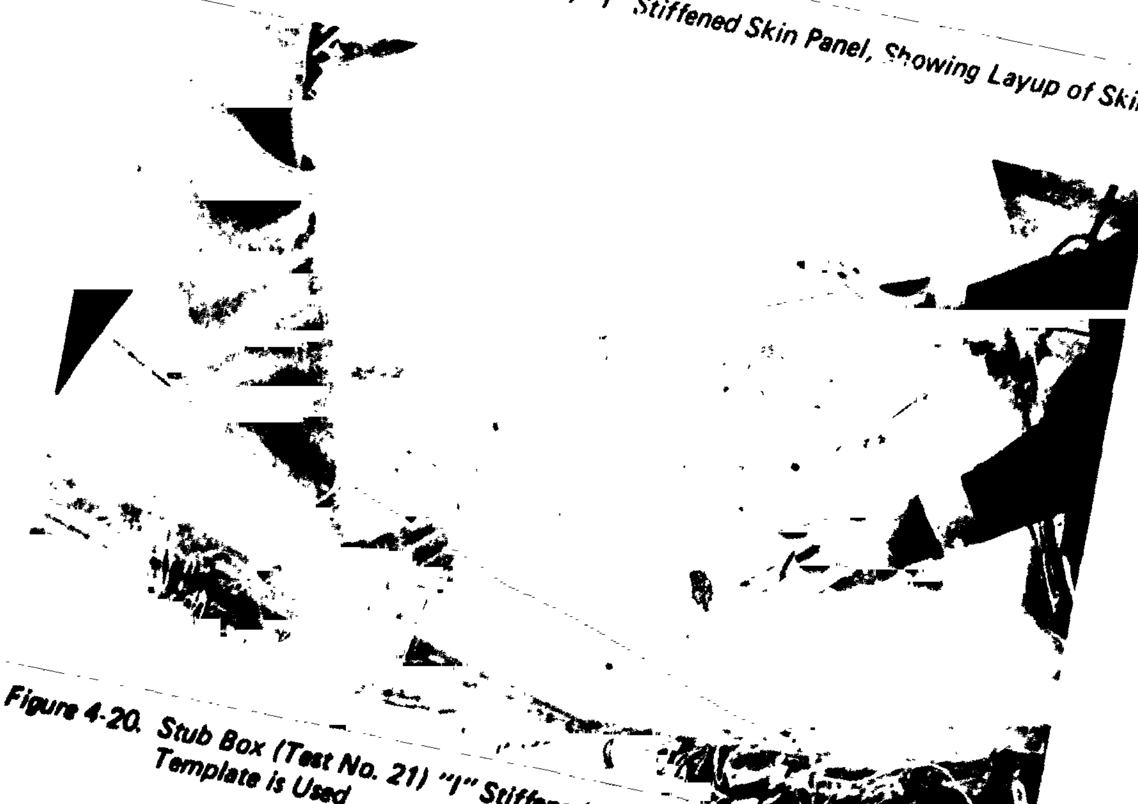

Figure 4-20. Stub Box (Test No. 21) "I" Stiffened Skin Panel, Showing How Locating Template is Used

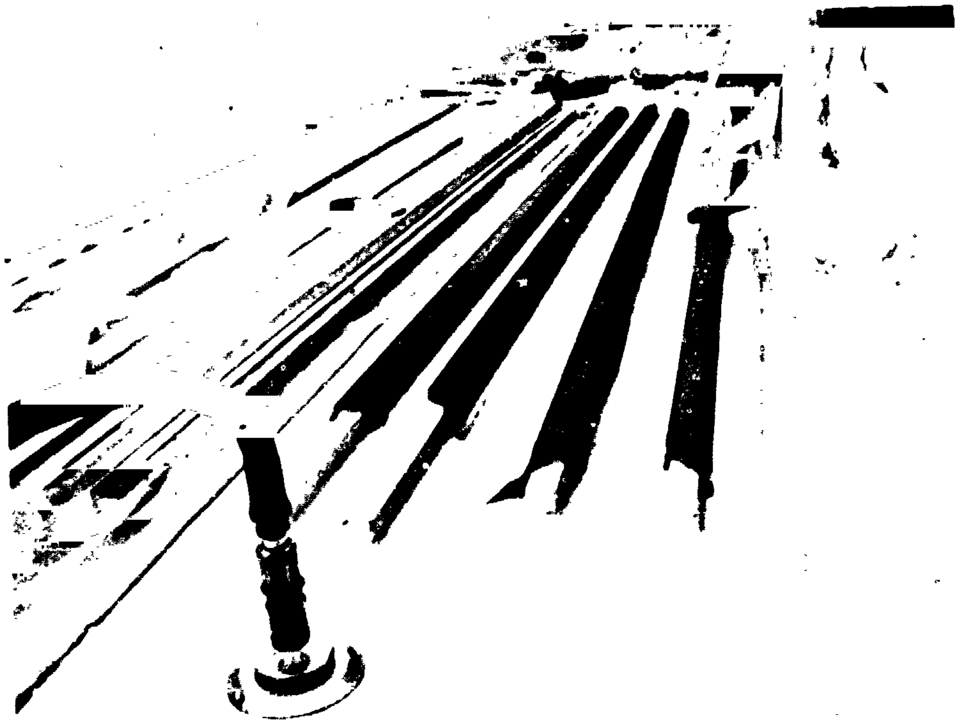

Figure 4-21. Stub Box (Test No. 21) "I" Stiffened Skin Panel, Showing Layup of "I" Stiffeners

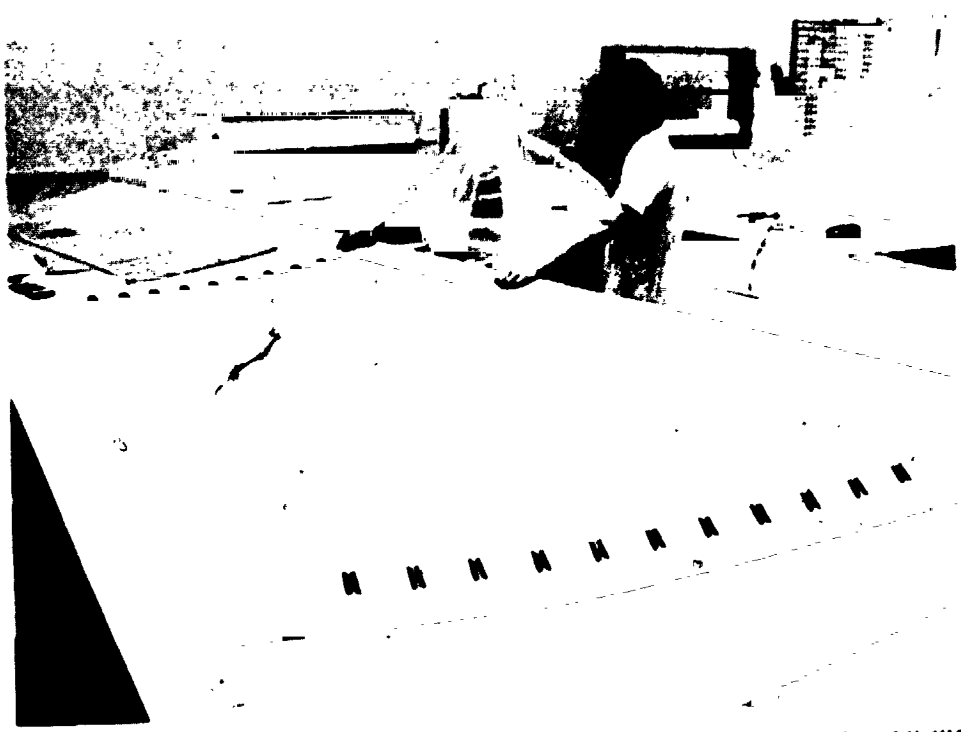

Figure 4-22. Stub Box (Test No. 21) "I" Stiffened Skin Panel, Showing All "I" Stiffeners in Place

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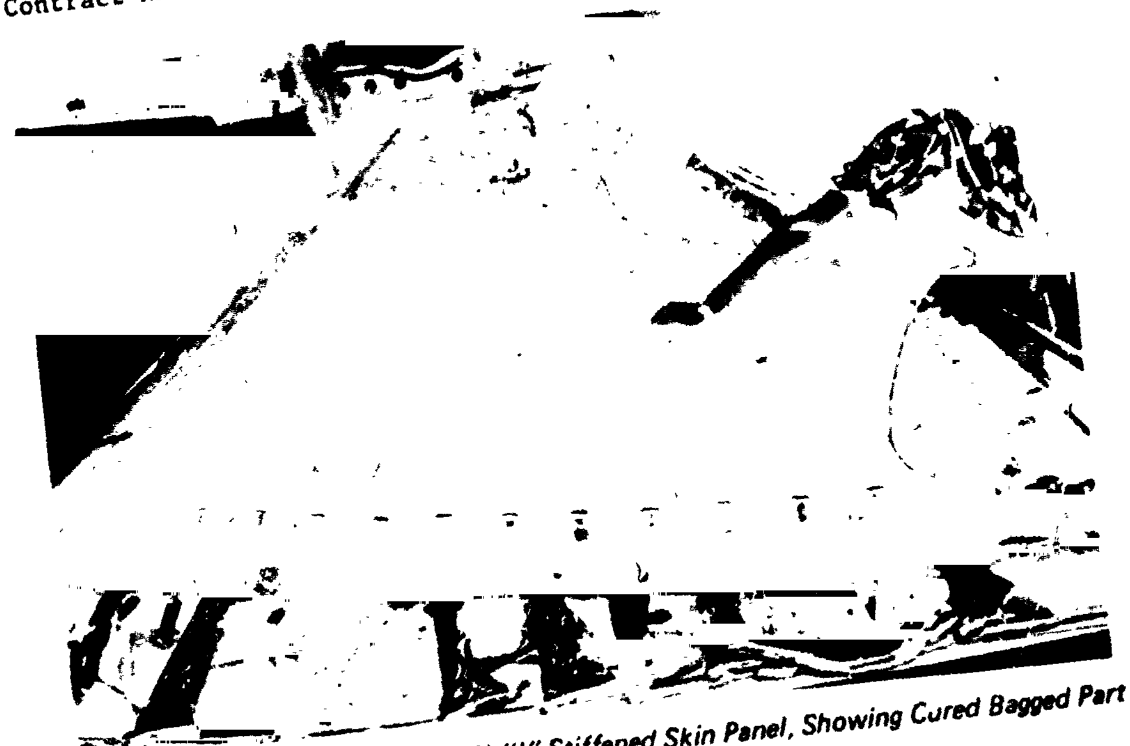

Figure 4-23. Stub Box (Test No. 21) "I" Stiffened Skin Panel, Showing Cured Bagged Part

Figure 4-24. Stub Box (Test No. 21) "I" Stiffened Skin Panel, Showing "I" Stiffened Side of Cured Panel

Figure 4-25. Stub Box (Test No. 21) "I" Stiffened Skin Panel, Showing Exterior Surface of Cured Panel

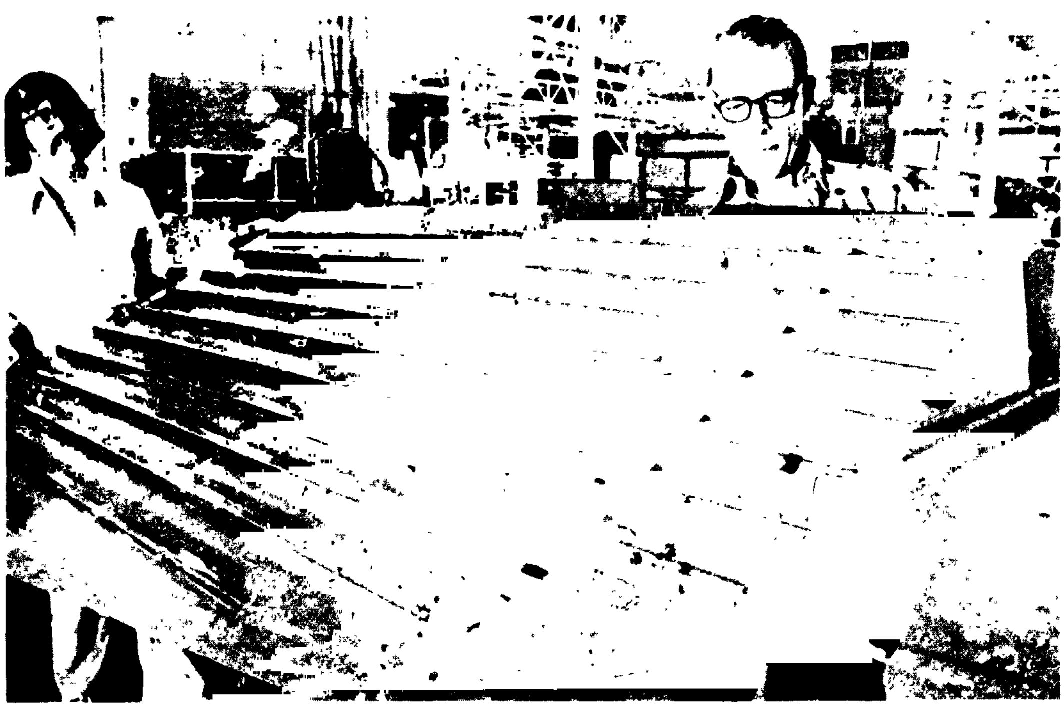

Figure 4-26. Stub Box (Test No. 21) "I" Stiffened Skin Panel, Showing Trimmed Part

Figure 4-27. Stub Box (Test No. 21), Showing Dummy Front Spar with Aluminum Nose Ribs

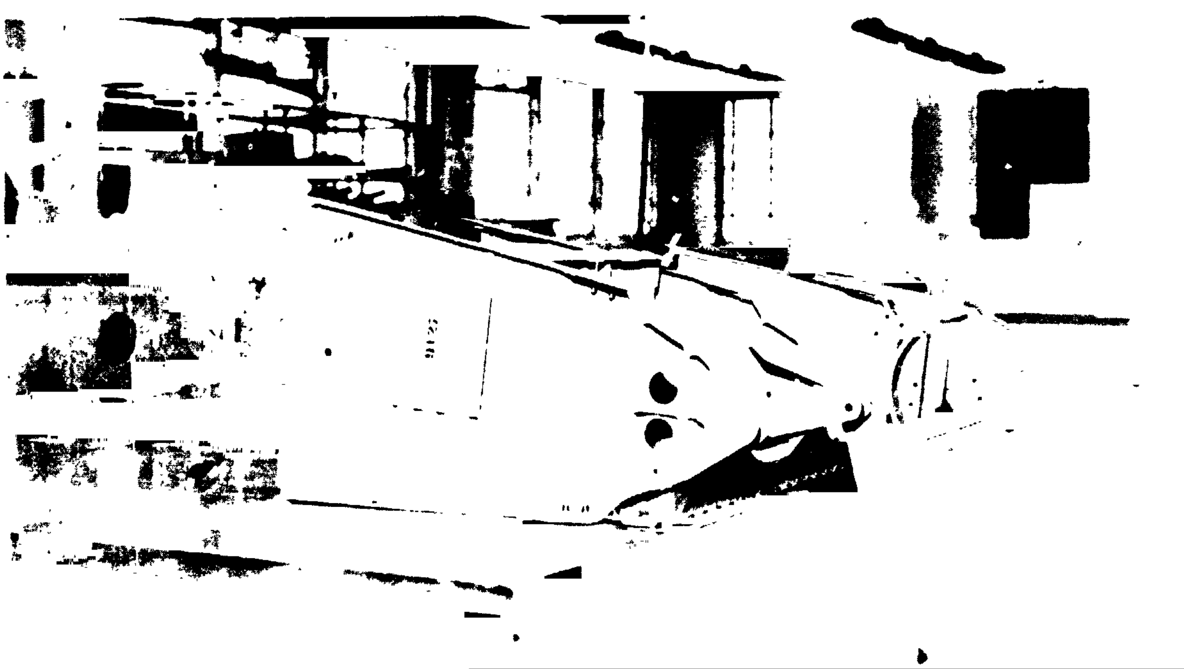

Figure 4-28. Stub Box (Test No. 21), Showing Aluminum Trailing-Edge Ribs and Fittings

SECTION 5.0

REFERENCES

1. "Advanced Composite Stabilizer for Boeing 737 Aircraft," First Quarterly Technical Progress Report, NASA Contract NAS1-15025, October 1977.

2. "Advanced Composite Stabilizer for Boeing 737 Aircraft," Second Quarterly Technical Progress Report, NASA Contract NAS1-15025, January 1978.

3. "A Standardized Load Sequence for Flight Simulation Tests on Transport Aircraft Wing Structure " NLR TR 73029 U, March 1973.

4. "Introduction to a Fighter Aircraft Loading Standard for Fatigue Evaluation," NLR MP 75017 U, May 20, 1975.

5. "Fatigue Strength of a Fiber Reinforced Material," D. Shutz and J.J. Gerharz, June 29, 1977.

"Advanced Composite Stabilizer for Boeing 737 Aircraft," Third Quarterly Technical Progress Report, NASA Contract NAS1-15025, April 1978.

7. "Thermo Physical Properties of Selected Aerospace Materials," Y. S. Touloukian and C. Y. Ho (Editors), Purdue University, TEPIAC/CINDAS, 1977.

APPENDIX A

ENGINEERING DRAWINGS

FOLDOUT FRAME

13 12 11

-82	-81	-75	-67	-66	-65	-64	-63	-62	-61	-60	-43	-42	-11	-10	-9	-8	-7	-6	-5	-4	-3	-2	-1	QTY REQD	
	✓																							6	BACB3
	✓																							6	BACN1
	✓																							6	AN960
													✓	✓	✓									6	BACB30
													✓	✓	✓									6	BACC30
													✓	✓	✓									6	AN960
				✓																	✓	✓	✓	16	BACB
				✓																	✓	✓	✓	16	BACC
				✓																	✓	✓	✓	16	AN960
	—																							—	
	✓																							1	
	-80																							1	
					✓																			1	
						✓	✓																		
					✓	✓																		2	
								✓	✓															3	
							✓																	2	
										✓														3	
								✓																1	
									✓	✓														1	
			—																					—	
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							—																	—	

13 12 11

FOLDOUT FRAME

11 10 9

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REQD	PART OR IDENTIFYING NUMBER	NOMENCLATURE OR DESCRIPTION	ZONE	MATERIAL AND SPECIFICATION		FINISH		REV LTR
6	BACB30NU3R4	BOLT	C6-4					
6	BACN10JC8CM	NUT	C6-4					
6	AN960C10L	WASHER	C6-4					
6	BACB30NY3 R4	BOLT	C6-4					
6	BACC30AB5S	NUT	C6-4					
6	AN960C8 L	WASHER	C6-4					
16	BACB30NY5 R4	BOLT	C4-2					
16	BACC30AB5S	NUT	C4-2					
16	AN960C8 L	WASHER	C4-2					
—	-81	PANEL ASSY, FATIGUE	A6-4		—	F-25.01	M	
1	-80	PANEL ASSY	C10-4		—			
1	-79	PANEL	C10-4	2 1	—			
1	-74	PANEL	A3-6	2 1	—			
1	-73	PANEL	A3-6	1	—			
2	-72	STRINGER	A4-6	2 1	—			
3	-72	STRINGER	A6-6	2 1	—			
2	-70	STRINGER	A4-6	1	—			
3	-70	STRINGER	A6-6	1	—			
1	-69	PANEL	A6-6	2 1	—			
1	-68	PANEL	A6-6	1	—	F 25.01		
—	-67	PANEL ASSY - SHEAR	A5-3	SAME AS -6 ASSY EXCEPT PAINT	—	19		
—	-66	PANEL ASSY - COMPRESSION	A6-2	SAME AS -2 ASSY EXCEPT PAINT	—	19		
—	-65	PANEL ASSY	A4-6		—	F-25.01		
—	-64	PANEL ASSY	A4-6		—	F-25.01		
—	-63	PANEL ASSY	A4-6		—	F-25.01	M	

LIST OF MATERIAL

-81	-67	-66	-65	-64	-63	-62	-61	-60	-43

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11 10 9

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-43	-42			-10	-9		-7	-6	-5		-3	-2	-1	QTY REQD	PART OR IDENTIFYING NUMBER	NOMENCLATURE OR DESCRIPTION	ZONE	MATERIAL AND SPECIFICATION
														—	-62	PANEL ASSY-CRIPPLING	A6-6	
														—	-61	PANEL ASSY-CRIPPLING	A6-6	
														—	-60	PANEL ASSY CRIPPLING	A6-6	
														—	-59	RAIL SHEAR SPECIMEN	B1-3	1 12
														—	-58	RAIL SHEAR SPECIMEN	B1-3	1
						✓	✓	✓						2	-56	GRIP-DOUBLER	B4-3	10 PLIES 14
														—	-51	RAIL SHEAR SPECIMEN 21	B1-3	2
✓	✓													14	-50	GRIP, UPPER-END	B6-5 C5-5	
	✓													2	-49	GRIP, UPPER-SIDE	C6-5 A6-5	
✓	✓													1	-48	GRIP, LOWER	A5-5	
														8	-47	STRINGER	A6-5	2
														8	-46	STRINGER	A6-5	2
														1	-45	PANEL	A6-5	
														1	-44	PANEL	A6-5	
—															-43	PANEL ASSY, COMP & SHEAR	A6-5	
	—													—	-42	PANEL ASSY, COMP & SHEAR	A6-5	
				✓										1	-40	PANEL ASSY	B10-4	
					✓									1	-39	PANEL ASSY	C10-4	
				-40											-38	PANEL	B10-4	22 2
					-39									1	-37	PANEL	C10-4	22 2
				-40	-39									4	-35	STRINGER	B9-4	2
				✓	✓									1	-33	RIB	A6-4	
				✓	✓									6	-32	OPP -31	A10-4	
				✓	✓									6	-31	DOUBLER	A10-4	
				✓	✓									4	-30	DOUBLER	A10-4	
							-28							7	-29	STRINGER	C7-3	
							✓							1	-28	PANEL ASSY	C7-3	
								-22						7	-27	STRINGER	B7-3	2
															-26			
								-22						1	-25	PANEL	A6-3	
							-28							1	-24	PANEL	A6-3	
								✓						1	-22	PANEL ASSY	B7-3	
									✓					1	-21	PANEL ASSY	C7-3	
											✓			5	-20	STRINGER	A6-2	
												✓	✓	5	-19	STRINGER	A6-2	2
									-21					1	-18	PANEL	A6-3	
									-21					5	-17	STRINGER	C6-3	2
												✓		1	-16	PANEL	A5-2	22 2
											✓	✓	✓	2	-15	RIB	A6-2 B7-2	
											✓		✓	1	-12	PANEL	A5-2	22 2
				—										—	-10	PANEL ASSY, FATIGUE	A6-4	
					—									—	-9	PANEL ASSY, FATIGUE	A6-4	
							—							—	-7	PANEL ASSY, SHEAR	A5-3	
								—						—	-6	PANEL ASSY, SHEAR	A5-3	
									—					—	-5	PANEL ASSY, SHEAR	A5-3	
											—			—	-3	PANEL ASSY, COMPRESSION	A6-2	
												—		—	-2	PANEL ASSY, COMPRESSION	A6-2	
													—	—	-1	PANEL ASSY, COMPRESSION	A6-2	

LIST OF MATERIAL

8 7

FOLDOUT FRAME

4 FOLDOUT FRAME

5 4 3

1 EPOXY PREIMPREGNATED GRAPHITE FABRIC PER BMS 8-212, TYPE II, CLASS 2, STYLE 3K-70-P FABRICATE PER BAC 5562 AND 10-61980. SEE PLY TABLES FOR APPLICATION

2 EPOXY PREIMPREGNATED GRAPHITE UNIDIRECTIONAL TAPE PER BMS 8-212, TYPE II, CLASS 1, GRADE 145. FABRICATE PER BAC 5562 AND 10-61980. SEE PLY TABLES FOR APPLICATION

3 PLY ORIENTATION CONVENTION:
FABRIC: 0° IS PARALLEL TO THE WARP DIRECTION
TAPE: 0° IS PARALLEL TO THE FIBER DIRECTION.

00 LIMITED SEE LIST OF MATERIAL FOR RELEASE.

4 BOND GRIP DOUBLER TO GRAPHITE/EPOXY PART WITH AMERICAN CYANAMID ADHESIVE FM 300K (.05LB/FT²), STORE AND CURE PER BAC 5514-5.04

5 HAND BLEND RADIUS AS SHOWN

6 WAVINESS IS NOT TO EXCEED .005 INCHES PER INCH OF SPECIMEN OR .050 INCHES OVER ENTIRE LENGTH OF SPECIMEN.

~~APPLY ONE COAT OF BMS 10-11, TYPE I PRIMER PER BAC 5736 ON ALL SIDES, EDGES & ENDS, OF SPECIMEN & END DETAIL AS SHOWN ON DRAWING~~

7 MACHINE TAPER OR DROP OFF PLIES EVENLY SPACED. MACHINED THICKNESS AT TAPER MAY BE A MAXIMUM OF .010;

8 PART MARKING LOCATION FOR ASSY NUMBER

9 EPOXY PREIMPREGNATED GRAPHITE UNIDIRECTIONAL TAPE PER BMS 8-212, TYPE II, CLASS 1, GRADE 95, 145 OR 190, AS REQUIRED. FABRICATE PER BAC 5562

10 TOLERANCE IS AS NOTED BY DECIMAL POINT .XX = ±.03, .XXX = ±.01

11 7075-T651 PLATE PER QQ-A-250/12

12 BLEND EPOXY RESIN FLASH SMOOTH WITH RESPECT TO ADJACENT GR/EP PARTS. USE 240-360 GRIT SANDPAPER.

13 BUTT SPLICE PER BAC 5562.

14 PREIMPREGNATED EPOXY RESIN GLASS FABRIC PER BMS 8-79, CLASS III, TYPE 1581 (.010 INCHES THICK PER CURED PLY) FABRICATE PER BAC 5470 (PEEL PLY REQUIRED ON BONDING SURFACES)

15 LAP SPLICE PER BAC 5562.

16 WARP DIRECTION FOR GRIP PLIES ONLY (-48, 49, -50, -56)

17 IN -35 P1 & P7 ARE ONLY 14.6 LONG & P2 & P8 ARE ONLY 13.0 LONG

18 EACH FOLLOWING GROUP OF GRAPHITE/EPOXY PARTS MUST BE MADE FROM THE SAME MATERIAL BATCH AND BE CURED IN THE SAME AUTOCLAVE LOAD:
58 WITH -24 59 WITH -25 AND -51 WITH -18
-70 68 AND -78/-73 WITH -20/-12
-72/-68 AND -72/-73 WITH -19/-12
-72/-69 AND 72/-74 WITH -19/-16

19 SRF 14672 + F-20.02 + F-14 3861-702 AS SPECIFIED ON DWG

20 -68 & -73 SAME EXCEPT WIDTH
-69 & -74 SAME EXCEPT WIDTH
-70 SAME AS -20 EXCEPT LENGTH
-33 SAME AS -13 EXCEPT LENGTH
-72 SAME AS -19 EXCEPT LENGTH

21 SEE EWA 22107 FOR FINAL MACHINING AND DRILLING INFORMATION FOR THE RAIL SHEAR SPECIMENS.

22 EPOXY PREIMPREGNATED GRAPHITE UNIDIRECTIONAL TAPE PER BMS 8-212, TYPE II, CLASS 1, GRADE 190. FABRICATE PER BAC 5562 AND 10-61980 SEE PLY TABLES FOR APPLICATION.

23 NO SPLICES ALLOWED

24 VOIDS AND POROSITIES ALLOWED IN BOND LINE IN UPPER FILLET AND FLANGE AND LOWER FILLET AREAS ONLY.

25 I.E., PLIES 1 THRU 9 = 45°; PLY 10 = 0°; PLIES 11 THRU 19 = 45°; PLY 20 = 0° ETC APPROX. 137 PLIES REQUIRED. INDICATED TOTAL THICKNESS MAY BE ACHIEVED BY MACHINING. OPTIONAL: LAYUP MAY BE MADE IN MULTIPLE STAGES OF THINNER PRECURED LAMINATES, STACKED AND BONDED PER 4 TO ACHIEVE THE FINISHED THICKNESS INDICATED

BREAK ALL SHARP CORNERS AND EDGES ON GRAPHITE DETAILS TO A RADIUS OR CHAMFER OF APPROXIMATELY .010. FINISHED PARTS SHALL HAVE SMOOTH CUT EDGES OF 125 RHR FINISH OR BETTER.

125 FINISH ON ALL MACHINED SURFACES.

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	F-25.01	M	
MATERIAL	MT-70	FINISH	PT-ASS

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4 3

5 FOLDOUT FRAME

2

...ITE DETAILS
...Y 010.
EDGES OF

FASTENER SYMBOL CODE

HOLE LOCATION FOR $\frac{1}{8}$ DIAMETER RIVET

HOLE LOCATION FOR $\frac{1}{4}$ DIAMETER BOLT

REVISIONS

SYM	ZONE	DESCRIPTION	DATE	APPROVAL

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A

TEST ONLY

APPLICATION

EWA 2211 OT

COMPOSITES

DRAWN O STRAIGHT

D WISS

THE BOEING COMPANY

NASA TEST SPECIMENS — SKIN PANEL TEST #10, GR/EP STABILIZER

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1 737 HORIZONTAL STABILIZER

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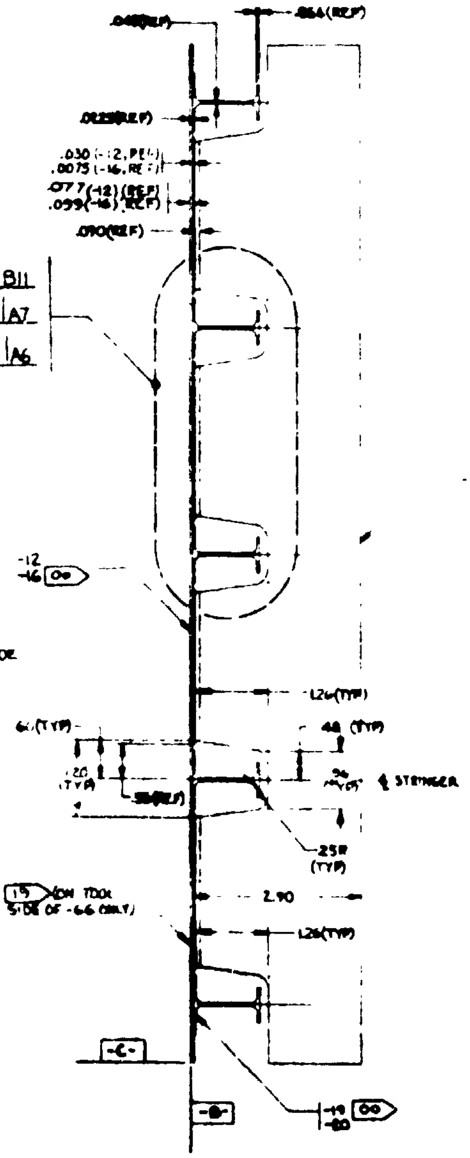
PLY TABLE II (-13)

PART NO	PLY NO "P"	MATERIAL	CLOTH WARP ORIENTATION	SPLICE	REV LTR
-13	1	1	0°	13	
	2		45°		
	3		0°		
	4		45°		
	5		0°		
	6		45°		
	7		45°		
	8		0°		
	9		45°		
	10		0°		
	11		45°		
	12	1	0°	13	



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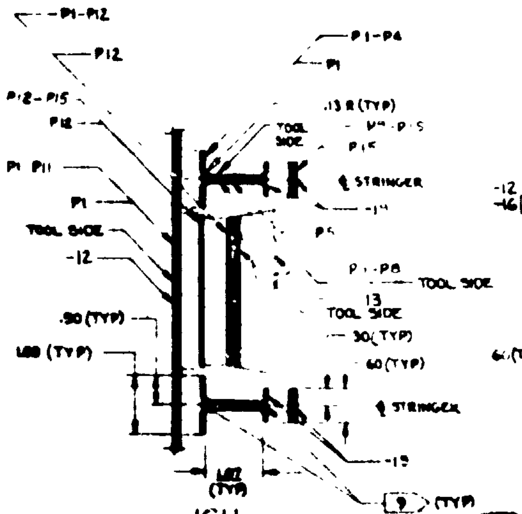


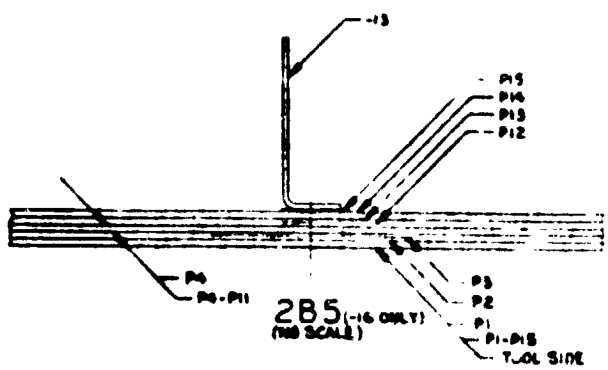
1C11

(-12, -13 & -19 ONLY) - NO SCALE
SEE PLY TABLES I & II - (EXPLODED VIEW)

IA10 A9

PLY TABLE I (-12)

PART NO	PLY NO "P"	MATERIAL	WARP ORIENTATION CLOTH WARP ORIENTATION	SPLICE	REV LTR
-12	1	1	45°	13	
	2	12	90°	21	
	3	12	90°	21	
	4	1	0°	15	
	5		45°		
	6		45°		
	7		45°		
	8	1	0°	14	
	[illegible]	[illegible]	[illegible]	[illegible]	
	[illegible]	[illegible]	[illegible]	[illegible]	
	[illegible]	[illegible]	[illegible]	[illegible]	
	[illegible]	[illegible]	[illegible]	[illegible]	
	13		0°		
	14		0°		
	15	1	45°	15	



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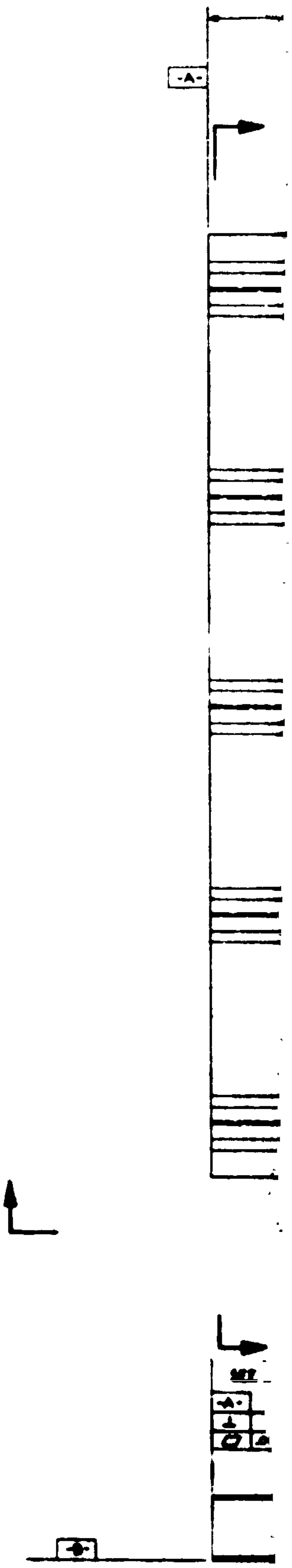

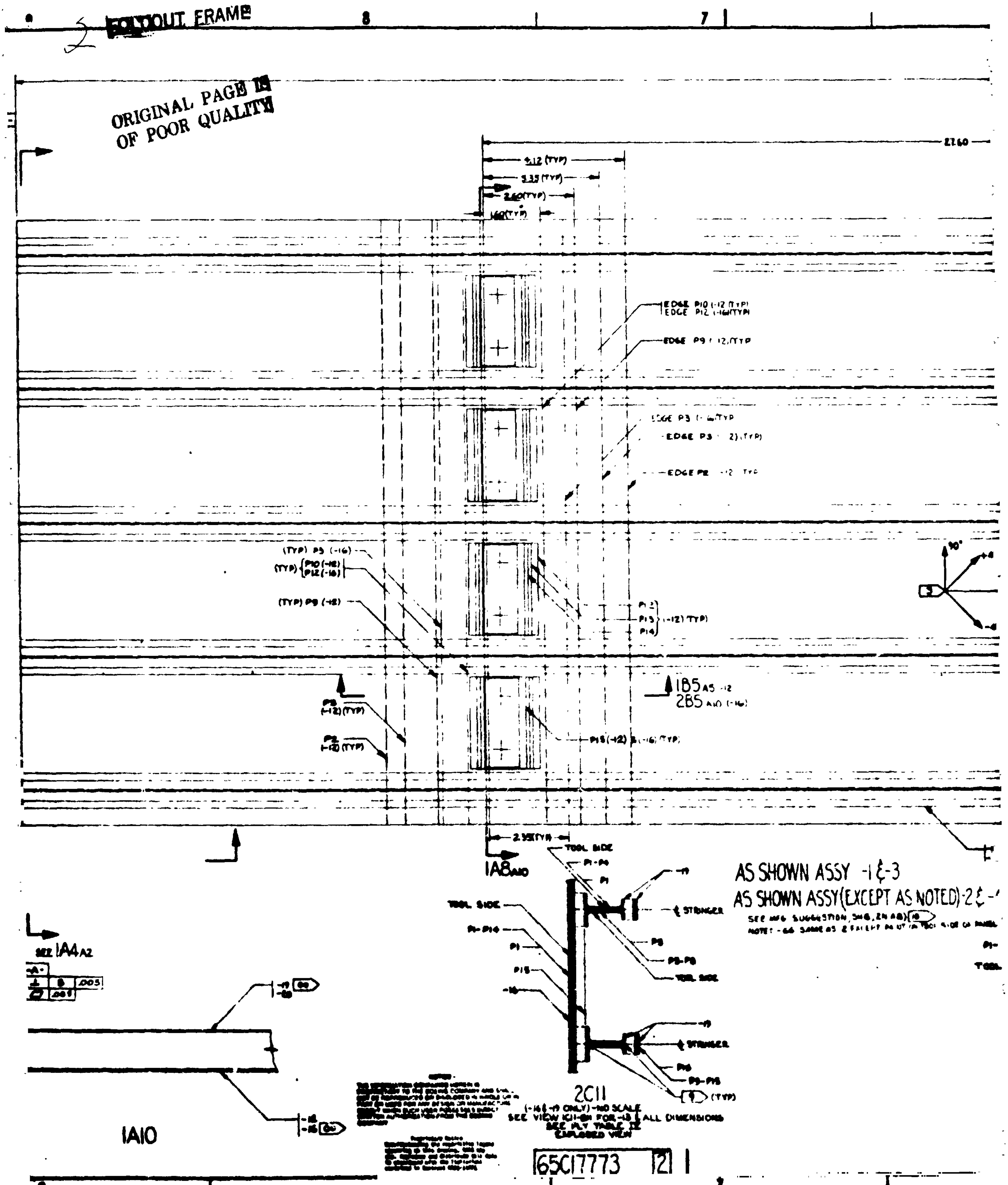
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8
7
27.60
4.12 (TYP)
3.35 (TYP)
2.60 (TYP)
1.60 (TYP)
(TYP) P3 (-16)
(TYP) P9 (-12)
P3 (-12) (TYP)
P2 (-12) (TYP)
1B5 A5 -12
2B5 A10 (-16)
2.35 (TYP)
TOOL SIDE
1A8 A10
TOOL SIDE
P1-P14
P1
P15
-16
STRINGER
P8
P8-P8
TOOL SIDE
STRINGER
P16
P9-P15
(TYP)
2C11
(-16 & -19 ONLY) - NO SCALE
SEE PLY TABLE
EXPLODED VIEW
AS SHOWN ASSY -1 & -3
AS SHOWN ASSY (EXCEPT AS NOTED) -2 &
SEE 1A4 A2
-A-
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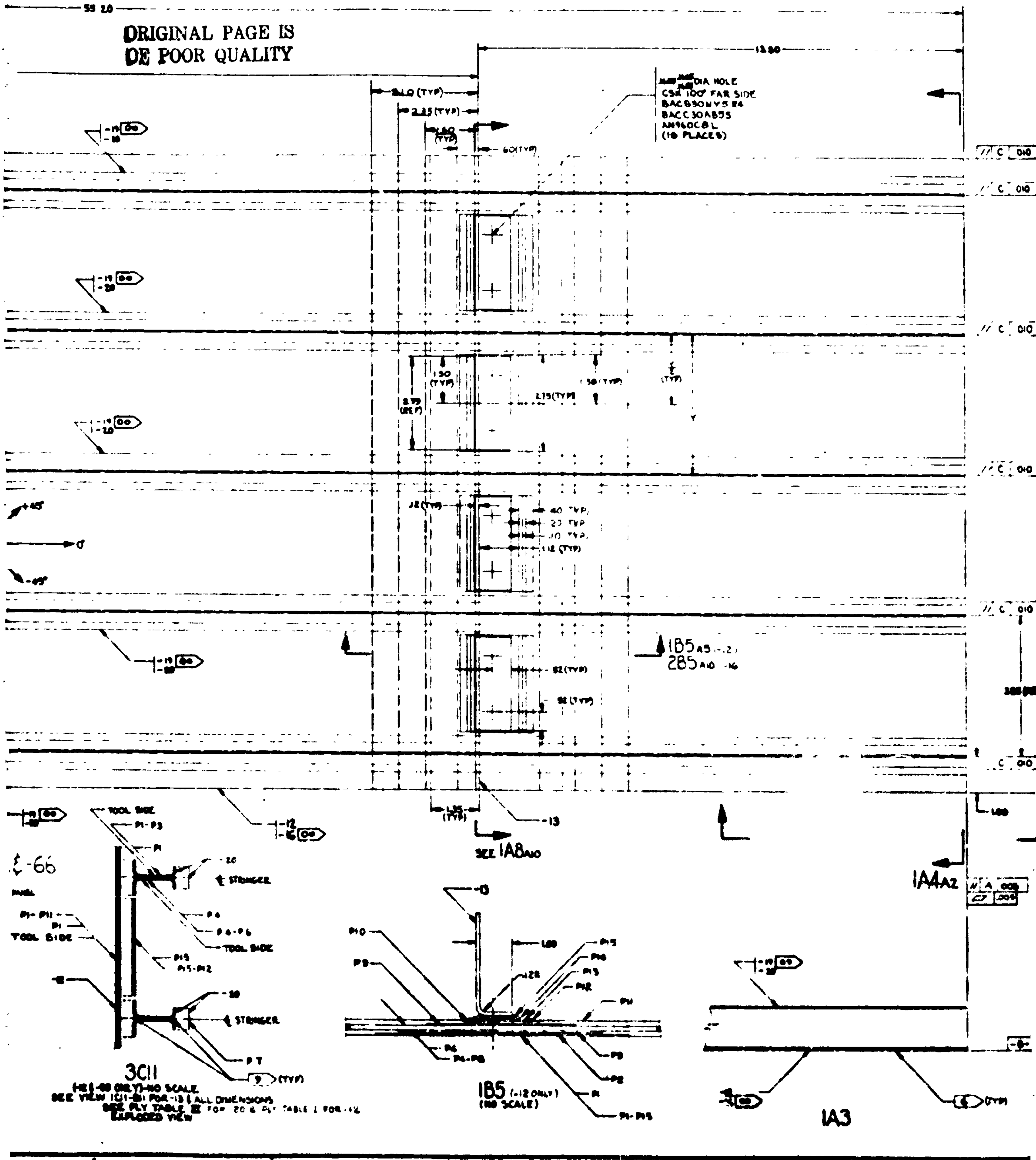

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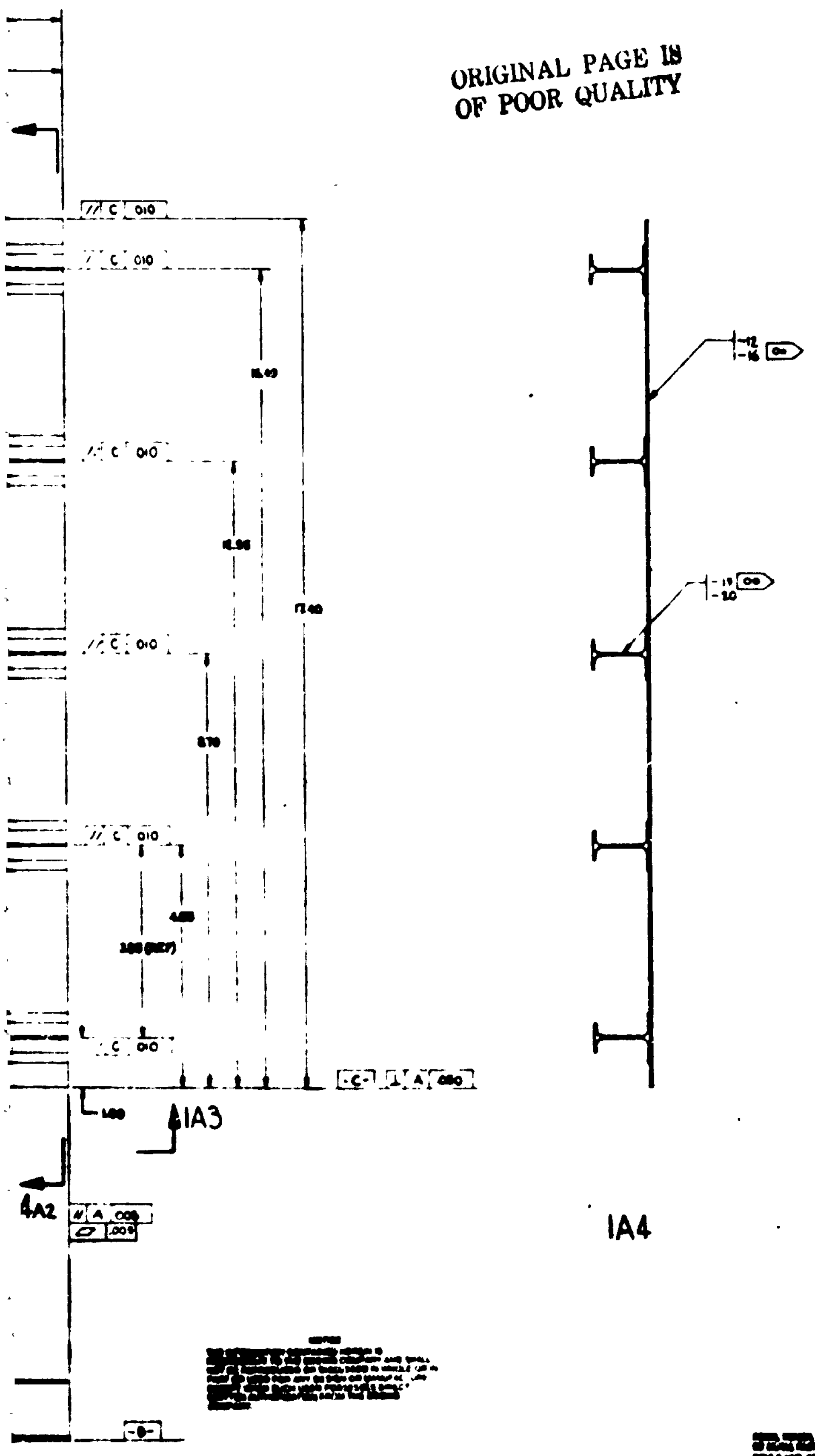

PLY TABLE III (-70 & -20 ONLY)

PART NO	PLY NO "P"	MATERIAL	TAPE ORIENTATION CLOTH WARP ORIENTATION	SPLICE	REV LTR
-20 & -70	1	1	45°	[illegible]	
	2		0°		
	3		45°		
	4		45°		
	5		0°		
	6		45°		
	7	1	45°	[illegible]	

PLY TABLE IV (-16, -19 & -72 ONLY)

PART NO	PLY NO "P"	MATERIAL	TAPE ORIENTATION CLOTH WARP ORIENTATION	SPLICE	REV LTR
-16	1	1	45°	[illegible]	
	2	22	90°	[illegible]	
	3	2	90°	[illegible]	
	4	1	0°	[illegible]	
	5		45°		
	6		45°		
	7		45°		
	8		0°		
	9		45°		
	10	1	45°		
	11	1	0°	[illegible]	
	12	2	90°	23	
	[illegible]	[illegible]	[illegible]	[illegible]	
	[illegible]		[illegible]		
	[illegible]		[illegible]		
	[illegible]		[illegible]		
-19 & -72	2		0°		
	3		0°		
	4		45°		
	5		45°		
	6		0°		
	7		0°		
	8	1	45°	[illegible]	
	9	2	0°	[illegible]	
	10		0°		
	11		0°		
	12		0°		
	13		0°		
	14	2	0°	[illegible]	
	15	1	45°	[illegible]	

TEST ONLY

NASA TEST S[illegible]
SKIN PANEL
GR/EP STAB[illegible]

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737 HORIZON[illegible]

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REVISIONS

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B

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A

-72 ONLY

A-3

THE BOEING COMPANY

NASA TEST SPECIMENS —
SKIN PANEL TEST #10,
GR/EP STABILIZER

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737 HORIZONTAL STABILIZER

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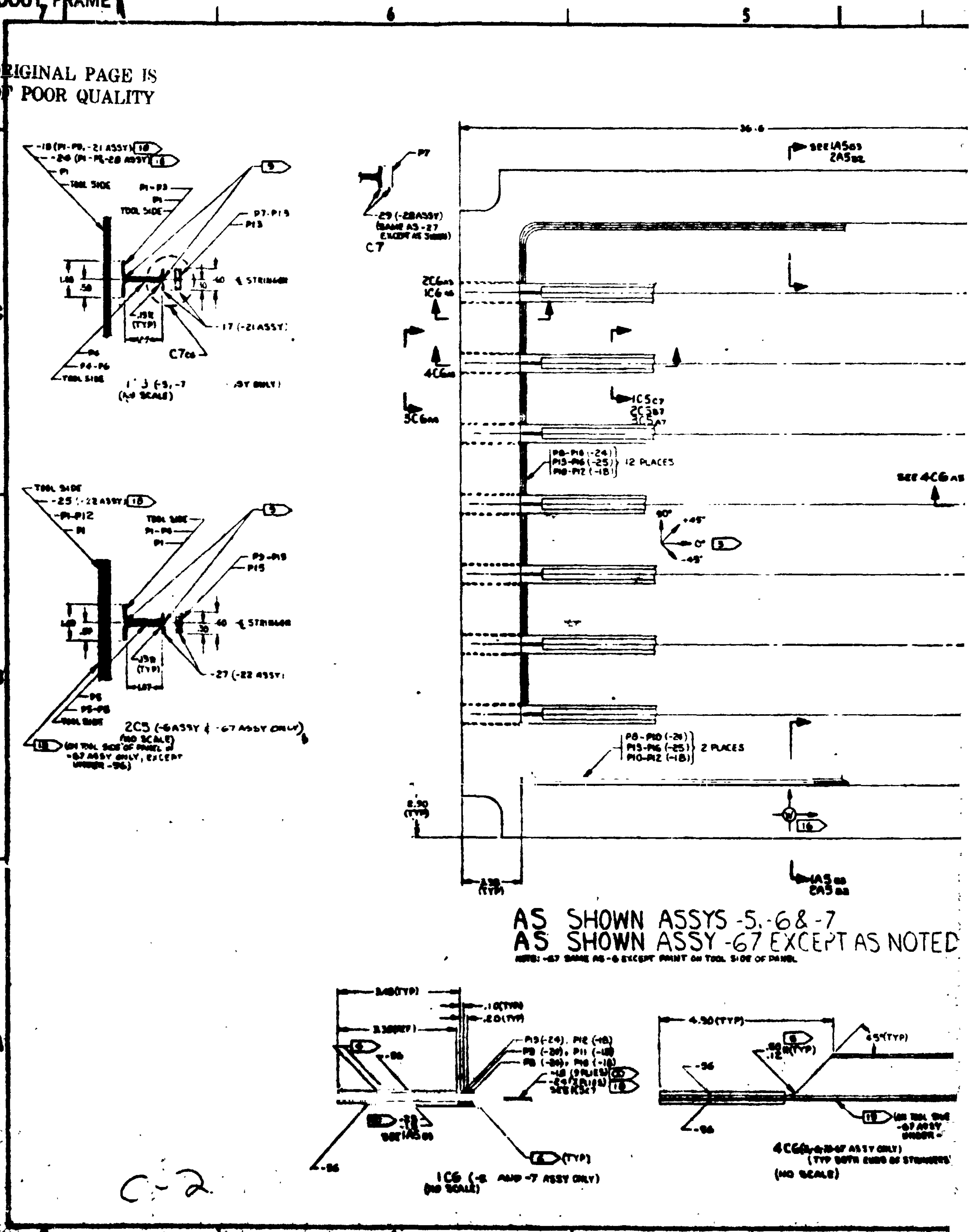

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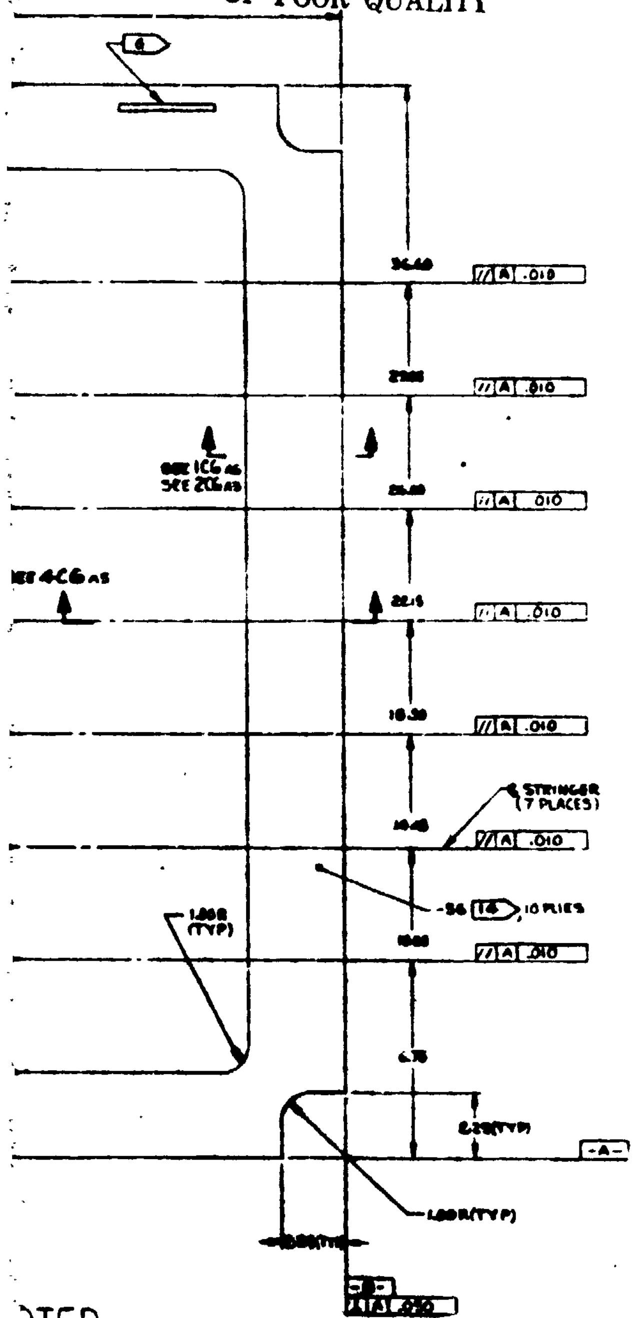
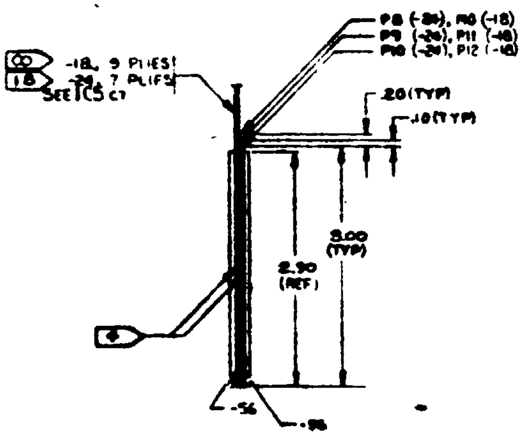
PLY TABLE					
PART NO	PLY NO "P"	MATERIAL	TAPE/CLOTH WARP ORIENTATION	SPLICE	REV LTR
-24	1,3,4,5,7,8,10	1	45°	15	
	2,6,9		0°		
-25	1,4,5,6,8,9,12,13,15,16	1	45°	15	
	3,7,10,14		0°		
	2,11	22	90°	23	
-27	1,4,5,8,15	1	45°	15	
	2,3,6,7		0°		
	9,10,11,12,13,14	2	0°	23	
-29	1,3,4,6,7	1	45°	15	
	2,5		0°		
-[illegible]	1,3,4,5,7	1	45°	15	
	2,6		0°		
-59	1,4,5,6,8,9,12	1	45°	15	
	3,7,10		0°		
	2,11	22	90°	23	
-18	1,4,5,6,9,10,12	1	45°	15	
	3,7,11	1	0°	15	
	2,8	22	90°	23	
-17	1,3,4,6,13	1	45°	15	
	2,5	1	0°	15	
	7,8,9,10,11,12	2	90°	23	
-51	1,4,5,6,9	1	45°	15	
	3,7	1	0°	15	
	2,8	22	90°	23	





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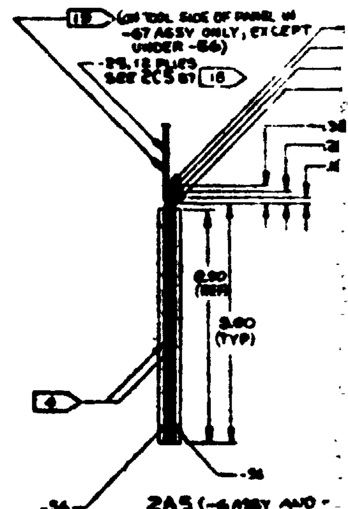

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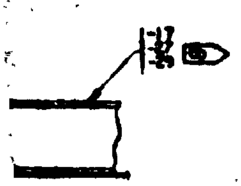
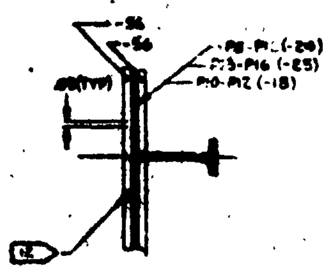
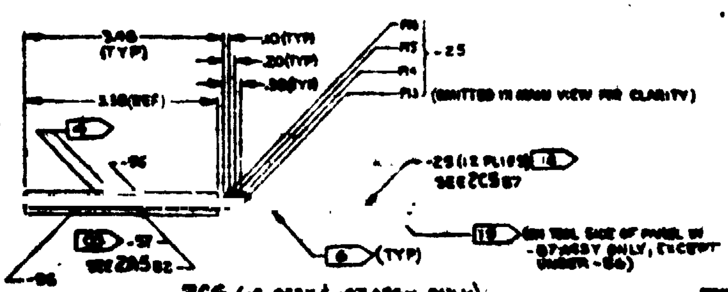
(NO SCALE)

OTED





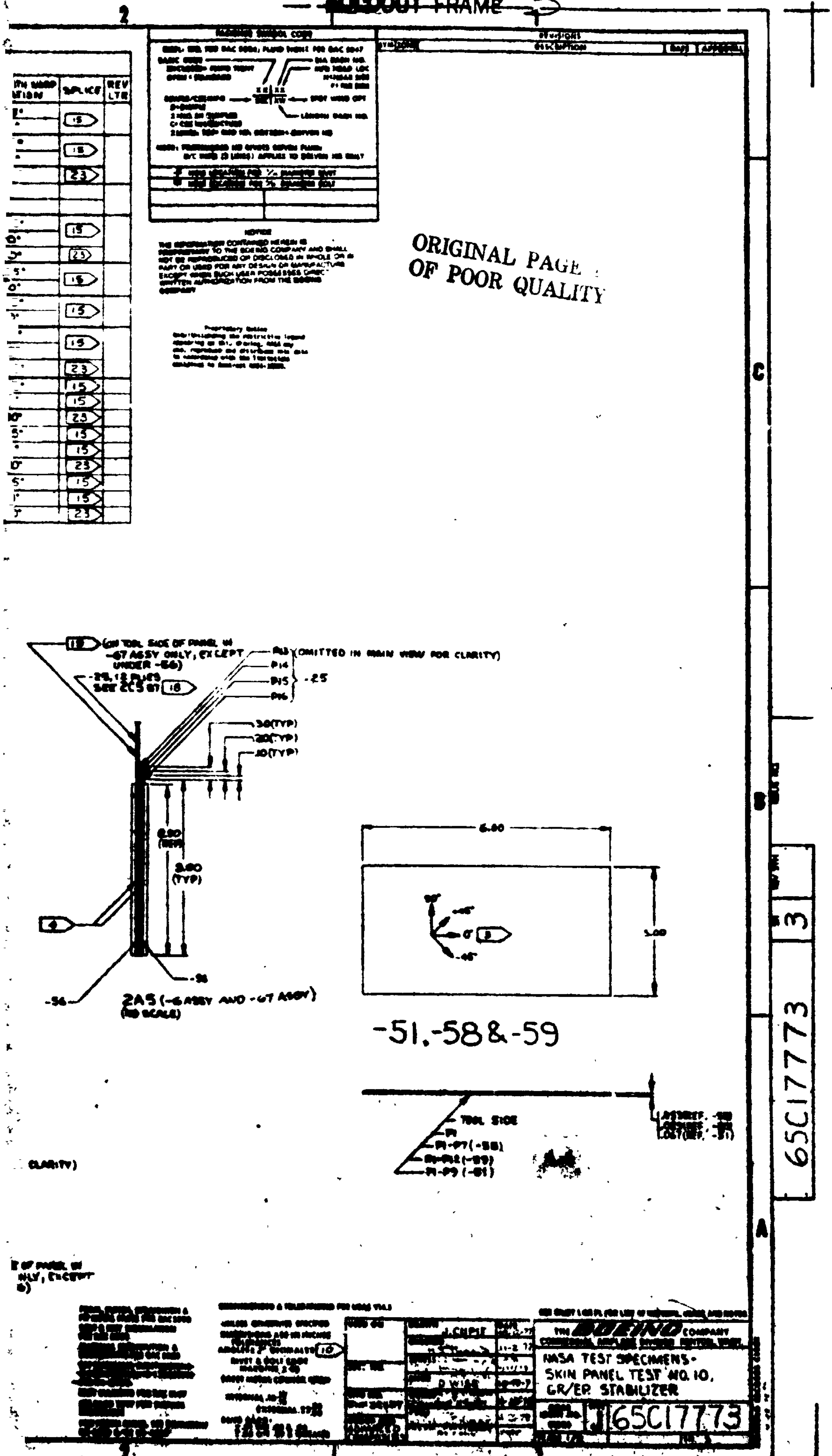
3C6
(NO SCALE)



2C6 (-6 ASSY & -67 ASSY ONLY)
(NO SCALE)

65C17773 3

FOLDOUT FRAME 3



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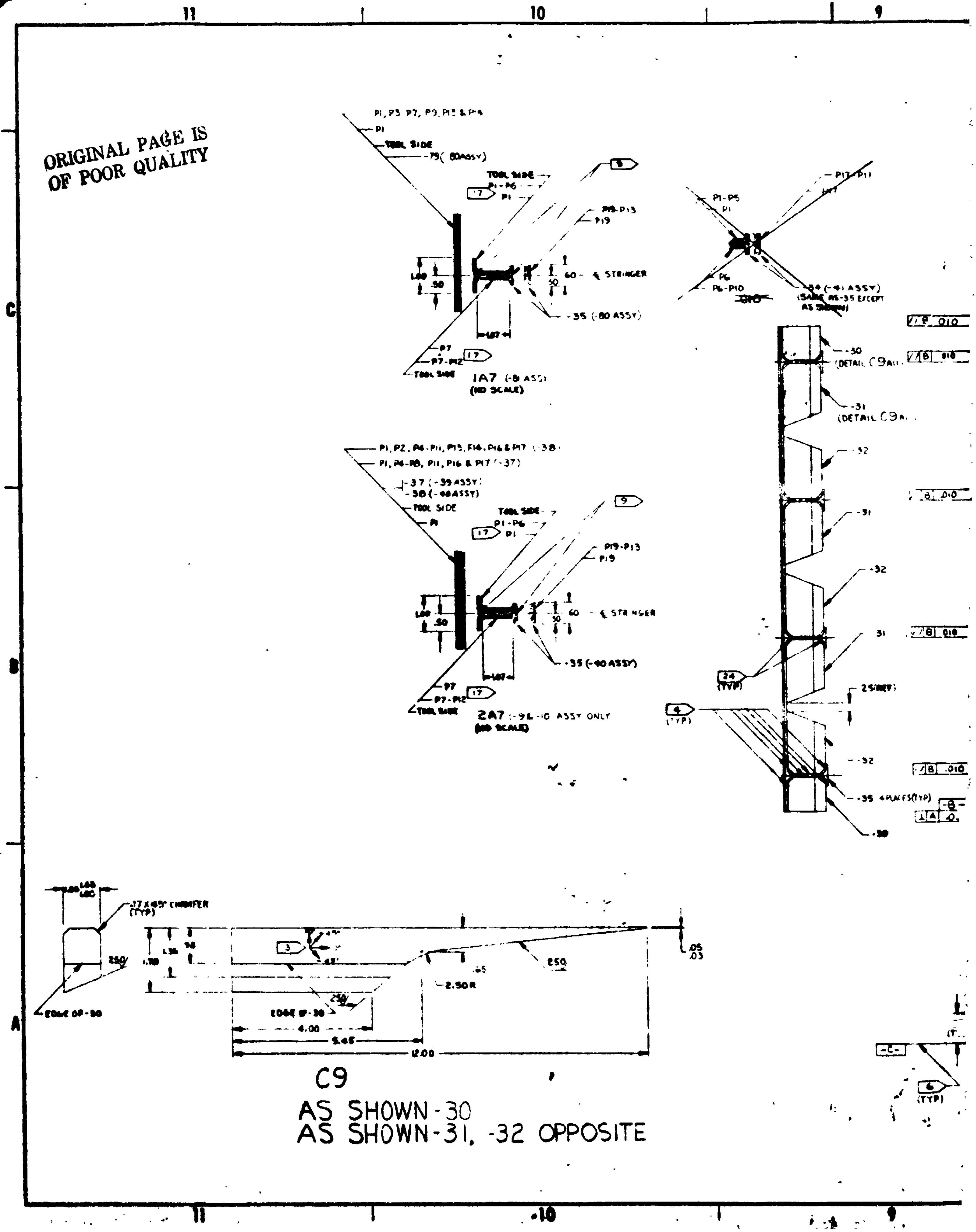
FOLDOUT FRAME



FOLDOUT FRAME 2

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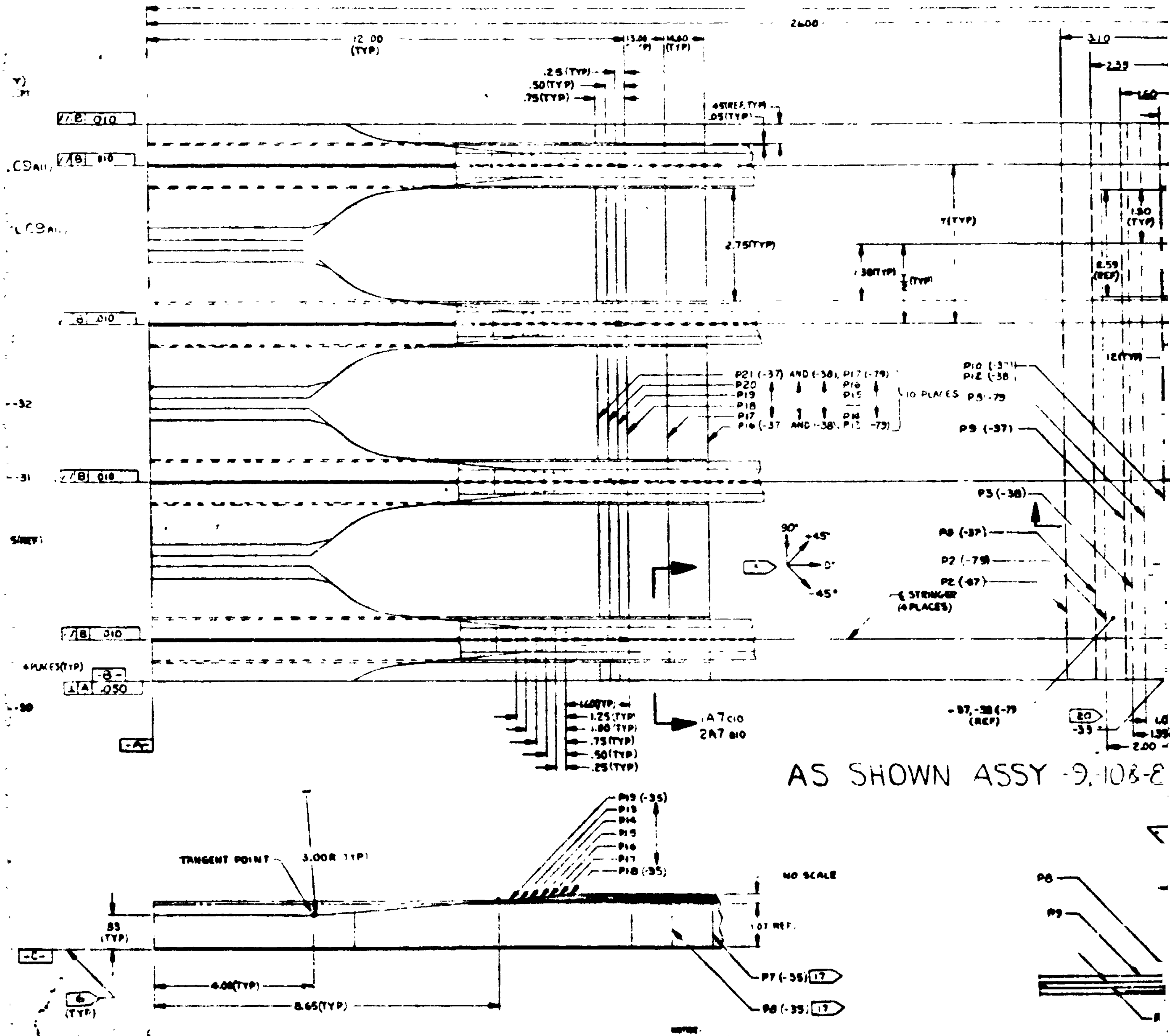

65C17773 4

6 FOLDOUT FRAME 3 5 4

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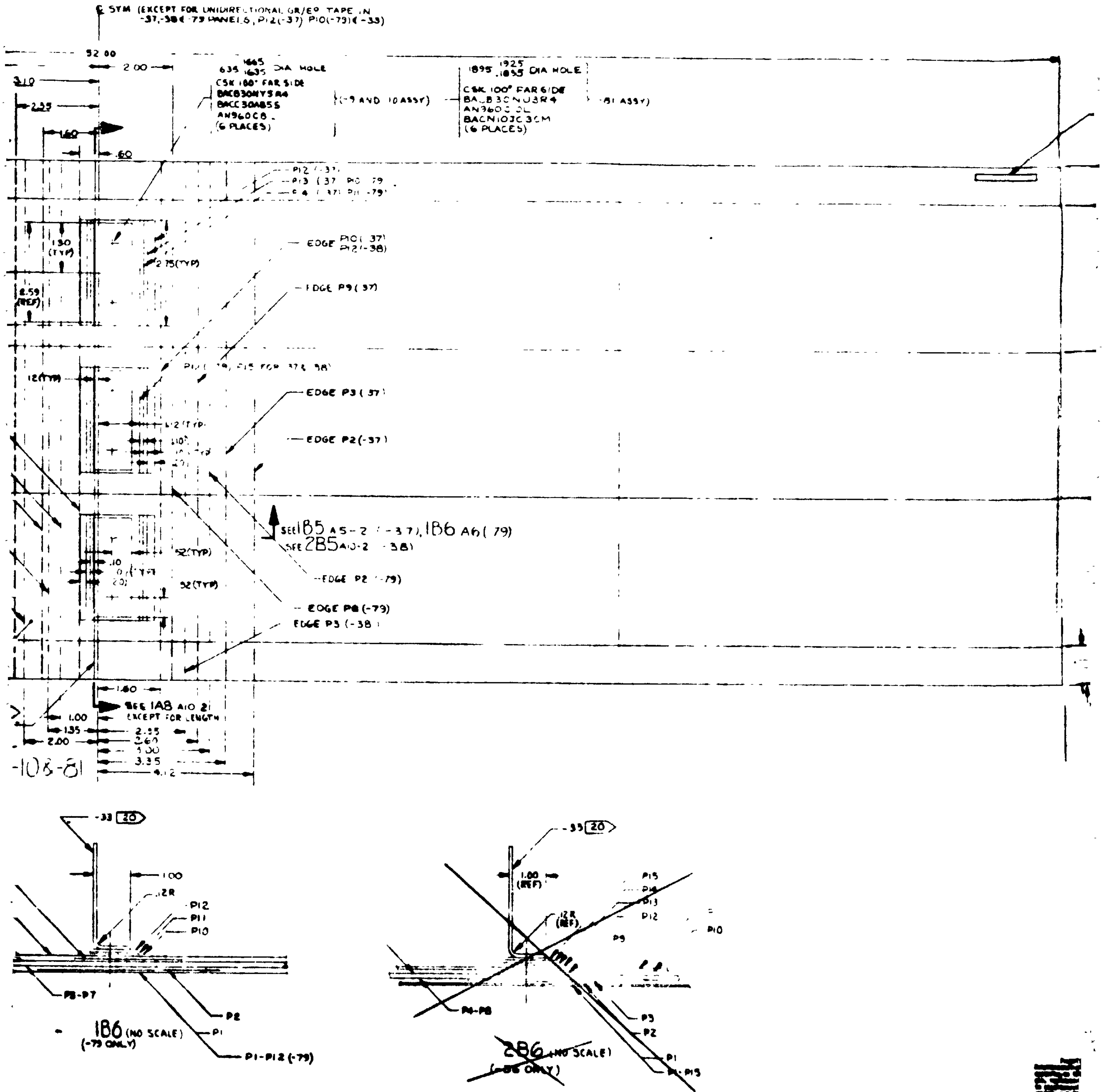

6 5 4

FOLDOUT FRAME 4

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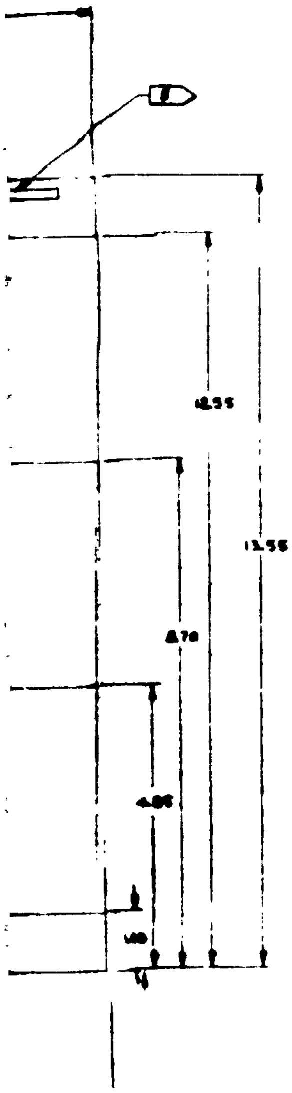
NOTICE

THE INFORMATION CONTAINED HEREIN IS PROPRIETARY TO THE BOEING COMPANY AND SHALL NOT BE REPRODUCED OR DISCLOSED IN WHOLE OR IN PART OR USED FOR ANY DESIGN OR MANUFACTURE EXCEPT WHEN SUCH USER POSSESSES DIRECT WRITTEN AUTHORIZATION FROM THE BOEING COMPANY

FASTENER SYMBOL CODE

HOLE LOCATION FOR ⅛ DIAMETER RIVET
HOLE LOCATION FOR ⅛ DIAMETER BOLT

PLY TABLE					
PART NO	PLY NO	MATERIAL	TAPE CLOTH WARP ORIENTATION	SPLICE	REV LTR
-33	1,3,5,8,10,12	1	0°	15	
	2,4,6,7,9,11		45°		
~~-34~~	~~1,3,5,6,8,10,17~~	~~1~~	~~45°~~	~~15~~	
	~~2,4,7,9~~		~~0°~~		
	~~11,12,13,14,15,16~~	~~2~~	~~0°~~	~~23~~	
-36	1,3,5,7,9,11,19	1	45°	15	
	2,4,6,8,10,12				
	13,14,15,16,17,18	2	0°	23	
-37	1,5,6,7,11,12,15,16,18	1	45°	15	
	4,8,13,14,17,19,20		0°		
	2,10	22	90°	23	
	3,9	2	90°		
-30, -31 & -32	90% OF PLIES 15		45°		
	10% OF PLIES (EVENLY SPACED THRU 45° PLIES)		0°	15	
-38	4,8,11,17,19,20		0°	15	
	1,5,6,7,9,10,14,15,16,18		45°		
	2,13	22	90°	23	
	3,12	2	90°		
-79	3,7,11,14,16		0°	15	
	1,4,5,6,9,10,12,13,15,17		45°		
	2,8	22	90°	23	
~~-36~~	~~1,5,6,7,11,12,15,16,18,2~~		~~45°~~		
	~~4,8,13,14,17,19,20~~				
	~~2,10~~	~~2~~		~~23~~	
	~~3,9~~	~~2~~	~~90°~~	~~23~~	



A-5

THE BOEING COMMERCIAL AIRPLANE

NASA TEST SKIN PANEL GR/EP

65C17773 4

FOLDOUT FRAME 5

ORIGINAL PAGE IS OF POOR QUALITY

CLOTH WARP ORIENTATION	SPLICE	REV LTR
0°	15	
45°		
45°	15	
0°		
0°	23	
45°	15	
0°		
0°	23	
45°	15	
0°		
90°	23	
90°		
45°	15	
0°		
0°		
45°	15	
90°		
90°	23	
0°		
45°	15	
90°	23	
45°		
0°		
90°	23	
90°	23	

THE BOEING COMPANY
COMMERCIAL AIRPLANE DIVISION RENTON, WASH

NASA TEST SPECIMENS
SKIN PANEL TEST NO 10,
GR/EP STABILIZER

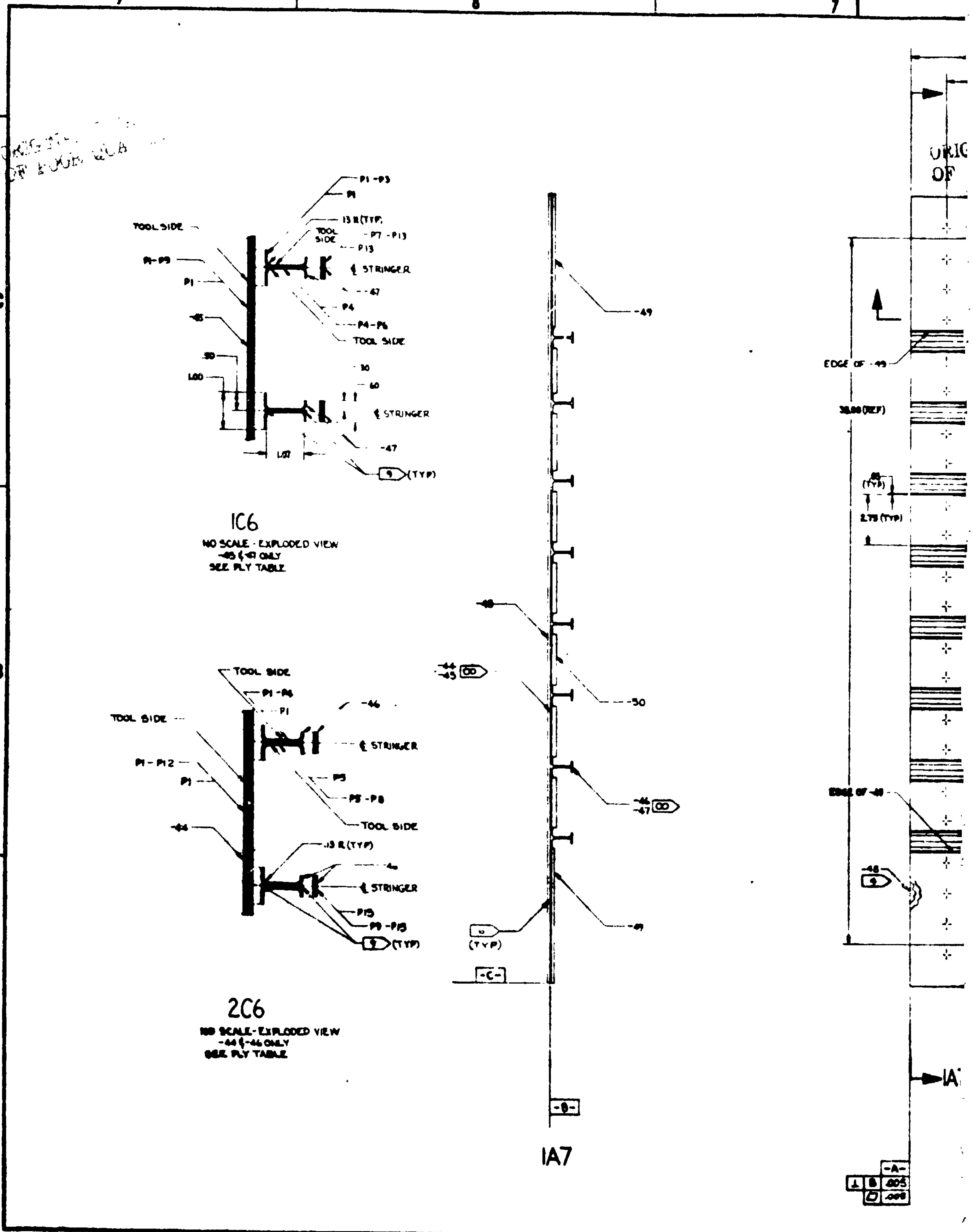
J 65C17773

SH 4

65C17773

65C17773 SH 4

FOLDOUT FRAME



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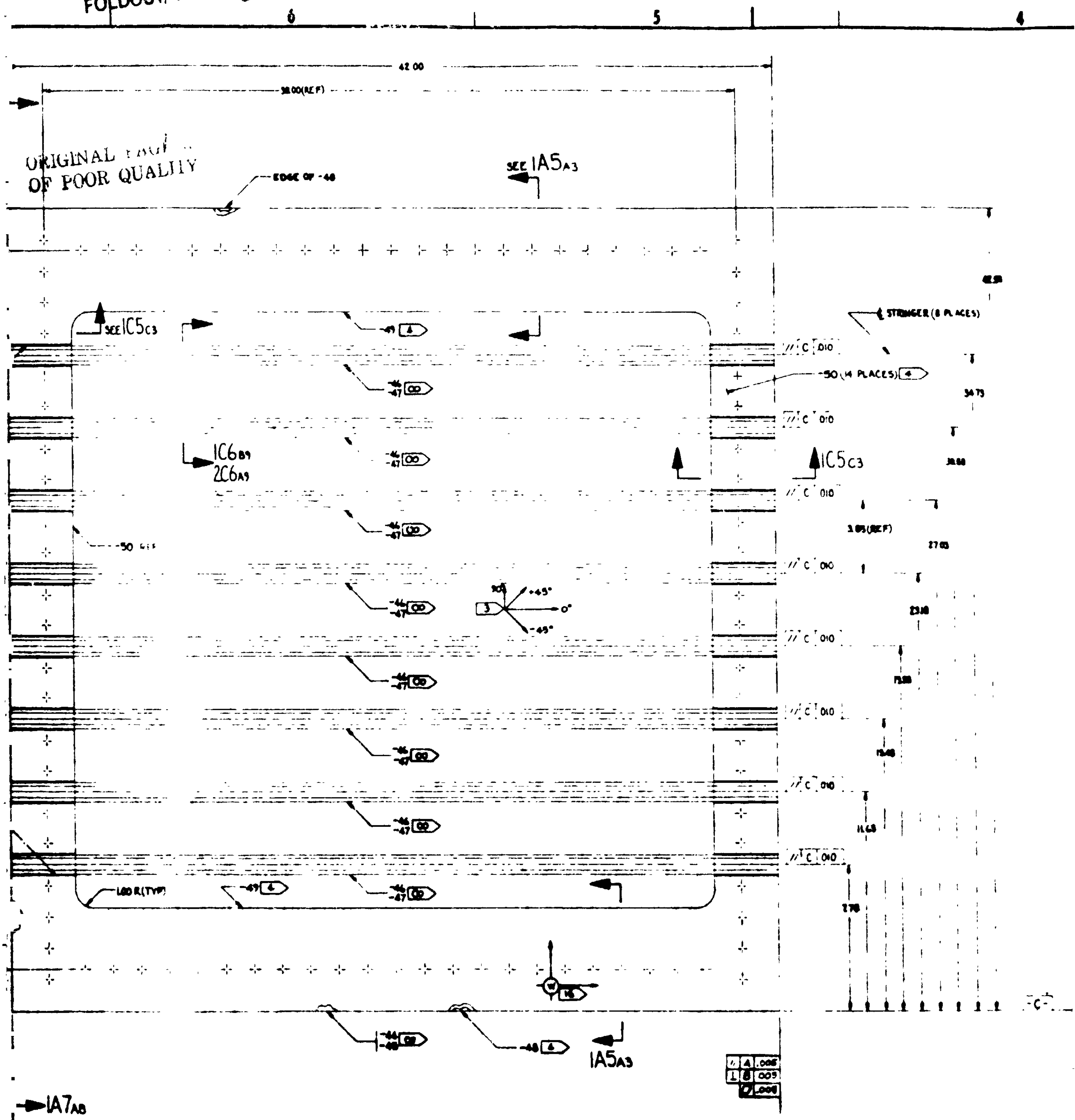
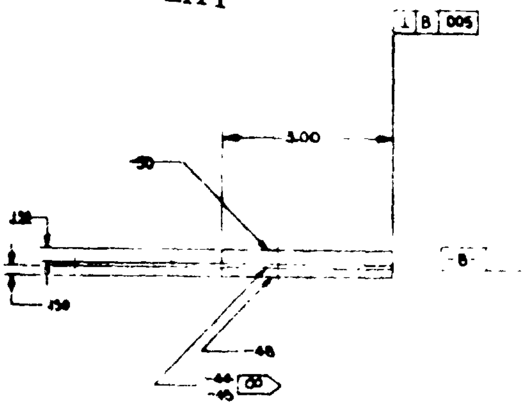


AS SHOWN ASSYS -42, -43

(-48, -49 & -50 ARE NOT DRAWN TO SCALE, THIS VIEW ONLY)

FOLDOUT FRAME

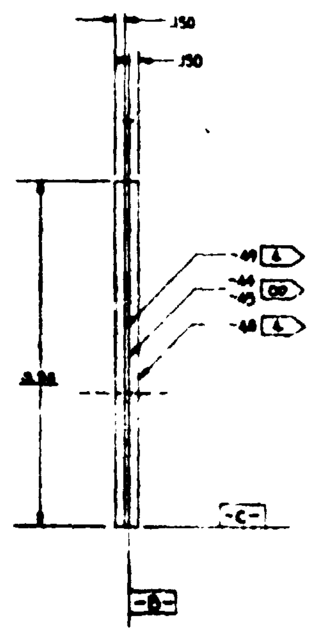
ORIGINAL PAGE IS OF POOR QUALITY



IC5
SCALE - 1/1

PLY TABLE

PART NUMBER	PLY NUMBER "P"	MATERIAL	CLOTH WARP ORIENTATION TAPE ORIENTATION	SPLICE	REV LTR
-45	1,4,5,6,9	1	45°	5	
	3,7	[illegible]	0°	[illegible]	
	2,8	[illegible]	[illegible]	3	
-47	1,3,4,6,13	1	45°	[illegible]	
	2,5	1	0°	15	
	7,8,9,10,11,12	2	0°	23	
-44	1,4,5,6,8,9,12	1	45°	15	
	3,7,10	[illegible]	0°	[illegible]	
	2,11	2	[illegible]	13	
-46	1,4,5,8,15	1	45°	15	
	2,3,6,7	1	0°	15	
	9,10,11,12,13,14	2	0°	23	



IA5
SCALE - 1/1

-C- ⊥ A .050

TEST ONLY

FOLDOUT FRAME 4

ORIGINAL PAGE IS
OF POOR QUALITY

SYMBOL CODE

REVISIONS

DESCRIPTION | DATE | APPROVAL

SPLICE	REV LTR
5	
3	
[illegible]	
15	
23	
19	
3	
15	
15	
23	

C

B

A

5

65C17773

THE BOEING COMPANY

TEST ONLY

STRAUGHT

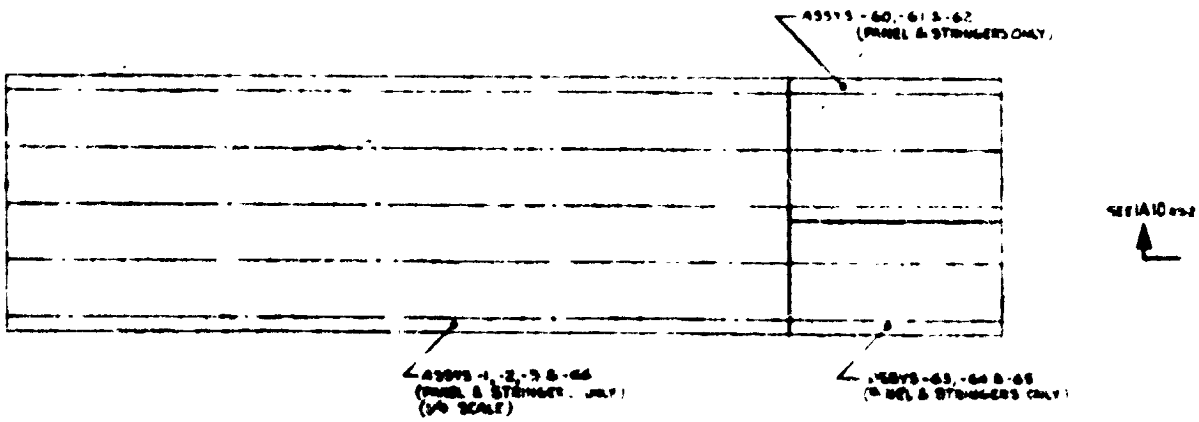
NASA TEST SPECIMENS —
SKIN PANEL TEST #10,
GR/EP STABILIZER

J 65C17773

737 HORIZONTAL STABILIZER

65C17773 SH 2

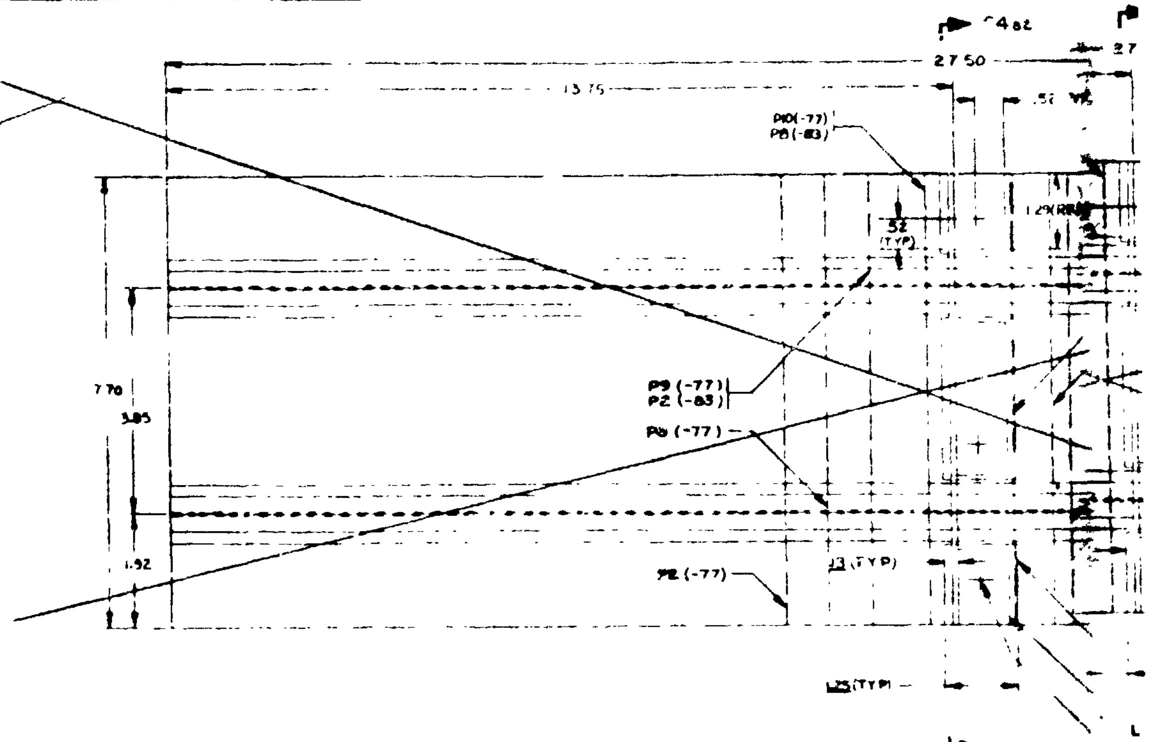
FOLDOUT FRAME



SUGGESTED METHOD OF MANUFACTURE:
CUT SPECIMENS FROM ONE [illegible] PANEL WITH STRINGERS
CAUTION: PROVIDE ADEQUATE EXCESS LENGTH AND WIDTH FOR TRIM

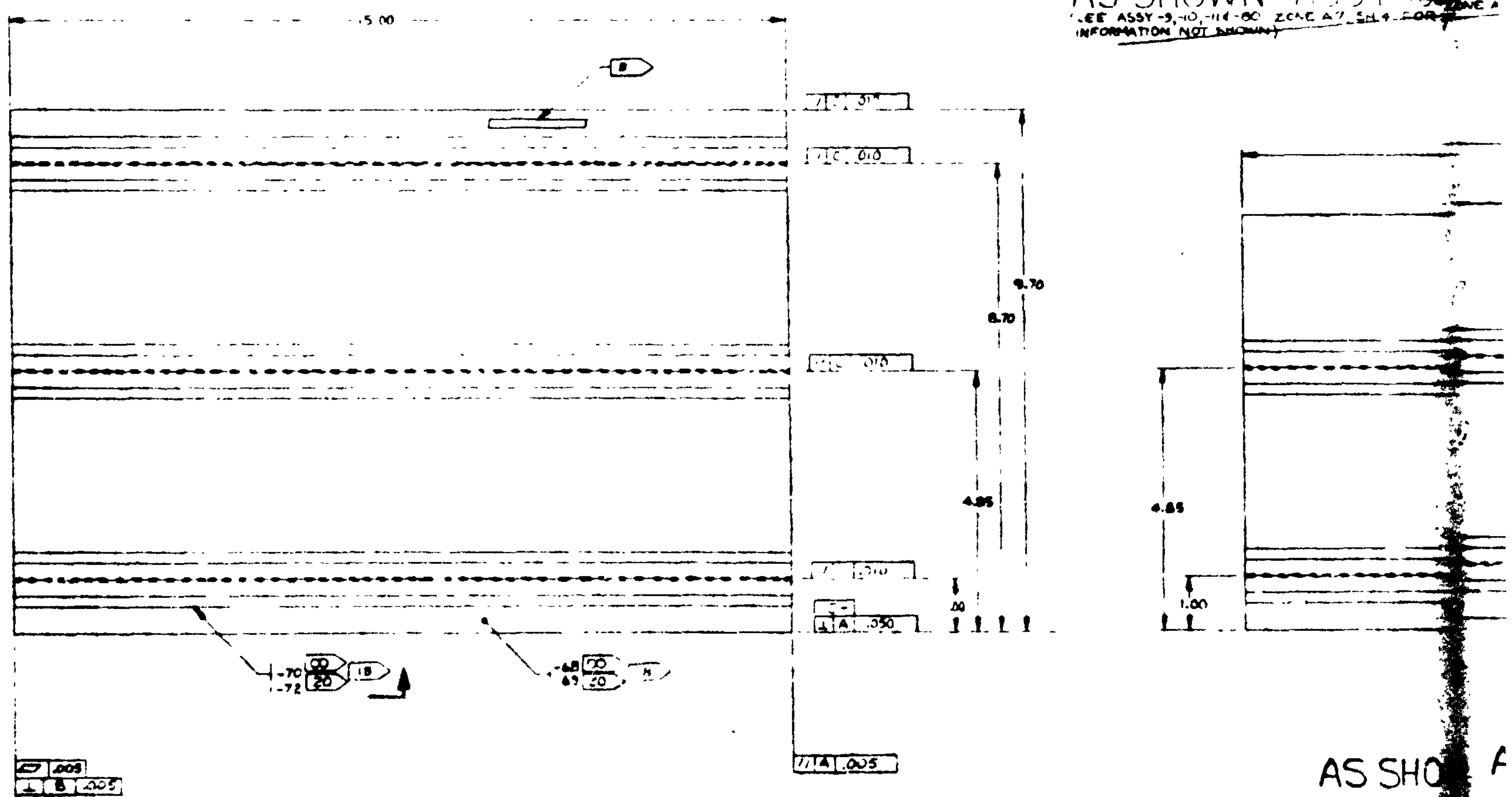
FOLDOUT FRAME

ORIGINAL PAGE IS
OF POOR QUALITY



AS SHOWN ASSY -56

(SEE ASSY -9, -10, -11, -60 ZONE A7, SH 4 FOR INFORMATION NOT SHOWN)



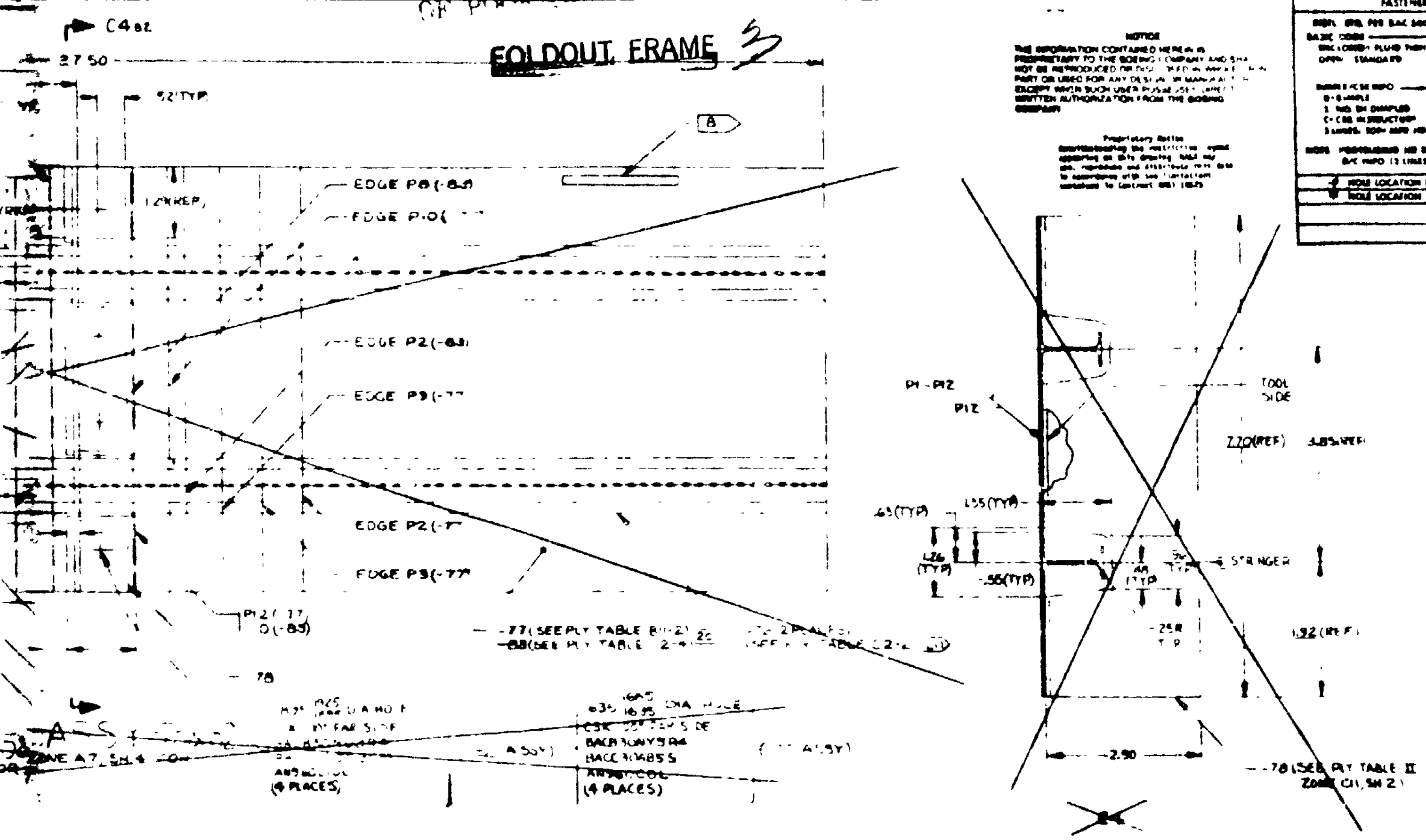
AS SHOWN ASSY -60, -61 & -62

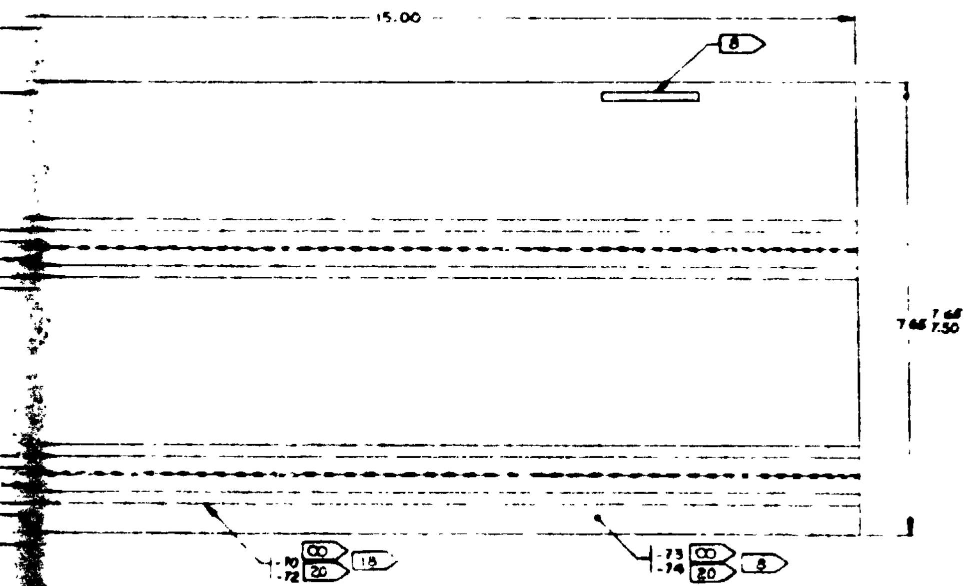
AS SHO

65C17773 6

ORIGINAL PAGE IS OF POOR QUALITY

FOLDOUT FRAME 3





PLY TABLE

PART NO	PLY NO "P"	MATERIAL	TAPE/CLOTH ORIENTATION
-69 & -74	1,4,5,6,8,9,12	1	45°
	3,7,10	1	0°
	2,11	22	
-68 & -73	1,3,4,5,7	1	
	2,6	1	

AS SHOWN ASSY -63, -6? & -65

FASTENER SYMBOL CODE

FOLDOUT FRAME

ORIGINAL PAGE IS OF POOR QUALITY

PLY TABLE

PLY NO "P"	MATERIAL	TAPE/CLOTH ORIENTATION	SPLICE	REV LTR
3,6,8,9,12	1	45°	15	
,10	1	0°	15	
[illegible]	22	90°	23	
,4,5,7	1	45°	15	
,6	1	0°	15	

C

B

A

6

65C17773

A-7

UNLESS OTHERWISE SPECIFIED
DIMENSIONS ARE IN INCHES
TOLERANCES
ANGLES ± DECIMALS ±
RIVET & BOLT EDGE MARGINS
SHEET METAL CORNER RADII
INTERNAL
EXTERNAL
BEND RADII

THE BOEING COMPANY
COMMERCIAL AIRPLANE DIVISION RENTON WASH

NASA TEST SPECIMENS -
SKIN PANEL TEST NO. 10
GR/EP STABILIZER

J 65C17773

SCALE 1/1

65C17773 SH 6

1. EPOXY PREIMPREGNATED GRAPHITE WOVEN FABRIC PER BMS 8-212, TYPE II, CLASS 2, STYLE 3K-70-P FABRICATE PER BAC 5562.
2. EPOXY PREIMPREGNATED GRAPHITE UNIDIRECTIONAL TAPE BMS 8-212, TYPE II CLASS 1, GRADE 145 FABRICATE PER BAC 5562
3. EPOXY PREIMPREGNATED GRAPHITE UNIDIRECTIONAL TAPE BMS 8-212, TYPE II, CLASS 1, GRADE 190. FABRICATE PER BAC 5562.
4. LAP SPLICE PER BAC 5562
5. PLY ORIENTATION CONVENTION
   FABRIC: 0° IS PARALLEL TO WARP DIRECTION
   TAPE: 0° IS PARALLEL TO FIBER DIRECTION
6. HONEYCOMB CORE, BMS 8-124, CLASS II, TYPE V GRADE 3.0 (1/8 IN CELL NOMEX)
7. CO-CURE SANDWICH WITH AMERICAN CYANAMID FM 300 BMS 8-245, TYPE II, CLASS 1, GRADE 03 ADHESIVE BETWEEN CORE & FACE SHEETS PER BAC 5562.
8. QUALITY CONTROL SHALL WEIGH THIS PART AND SUBMIT RECORDED WEIGHT IN WRITING TO G NISHIMURA 8 8091 M/S 30 01
9. NO SPLICES ALLOWED
10. BUTT SPLICE PER BAC 5562.
11. DROP OFF PLIES AT EQUAL SPACING WITHIN THE DIMENSION SHOWN
12. GRAPHITE FILLERS AS REQUIRED, EPOXY IMPREGNATED GRAPHITE UNIDIRECTIONAL TAPE, TYPE II, CLASS 1, GRADE 95, 145 OR 190 OPTIONAL, PER BMS 8-212. FABRICATE PER BAC 5562
13. THE LOCATION OF THE ℄ OF THE STIFFENERS MAY VARY ± .05 FROM THE BASIC POSITION. PERPENDICULARITY TO SKIN TOOL SURFACE SHALL BE WITHIN ± 2°
14. TOLERANCE ON PLY EDGE LOCATION ± .06
15. RUBBER STAMP
16. HONEYCOMB CORE BMS 8-124 CLASS II TYPE V, GRADE 3.0 (1/8 INCH CELL, NOMEX)
17. CAUTION: THIS PART IS COCURED WITH THE SKIN LAYUP DO NOT CURE THIS PART INDIVIDUALLY
18. PLY NO'S 27, 41, 42, 47 & 51 NOT USED
19. BACB30MY6R3 BOLT AND 24
20. BACB30MY6R5 BOLT AND 24
21. BACB30NW6R3 BOLT AND 25
22. BACB30NW6R5 BOLT AND 25
23. FOR -2 PANEL: BACB30NW3R3 BOLT AND 27
    FOR -3 PANEL: BACB30NY5R4 BOLT AND 26
24. BOLT HEAD ON -2 OR -3 SIDE
    AN960C10L WASHER
    BACC30M6 COLLAR
25. CSK 100° ON -2 OR -3 SIDE
    AN960C10L WASHER
    BACC30M6 COLLAR
26. CSK 100° -3 SIDE
    AN960C8L WASHER
    BACC30M5 COLLAR
27. CSK 100° -2 SIDE AND 28
28. BACN10KB3CM NUTPLATE
29. 2024-T3 PER QQ-A-250/4
30. IF GAP BETWEEN GR/EP RIB AND ALUM SPAR CHORD EXCEEDS .010 PRIOR TO FASTENER CLAMP UP, SHIM WITH BACS40R10C14F LAMINATED SHIM
31. IF GAP BETWEEN ALUM RIB AND GR/EP SKIN PANEL ASSEMBLY EXCEEDS .010 PRIOR TO FASTENER CLAMP UP, SHIM WITH BACS40R10D30F LAMINATED SHIM
32. PLY NO'S. 27, 26, 31 & 33 NOT USED

FOLDOUT FRAME

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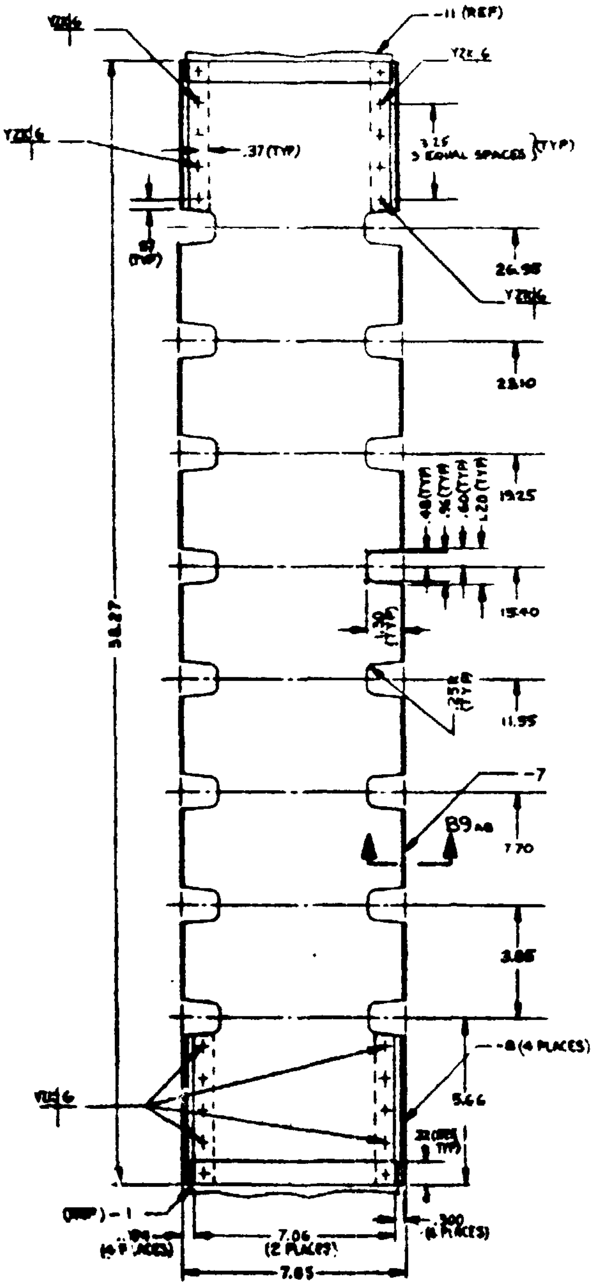
11 10 9

FOLDOUT FRAME



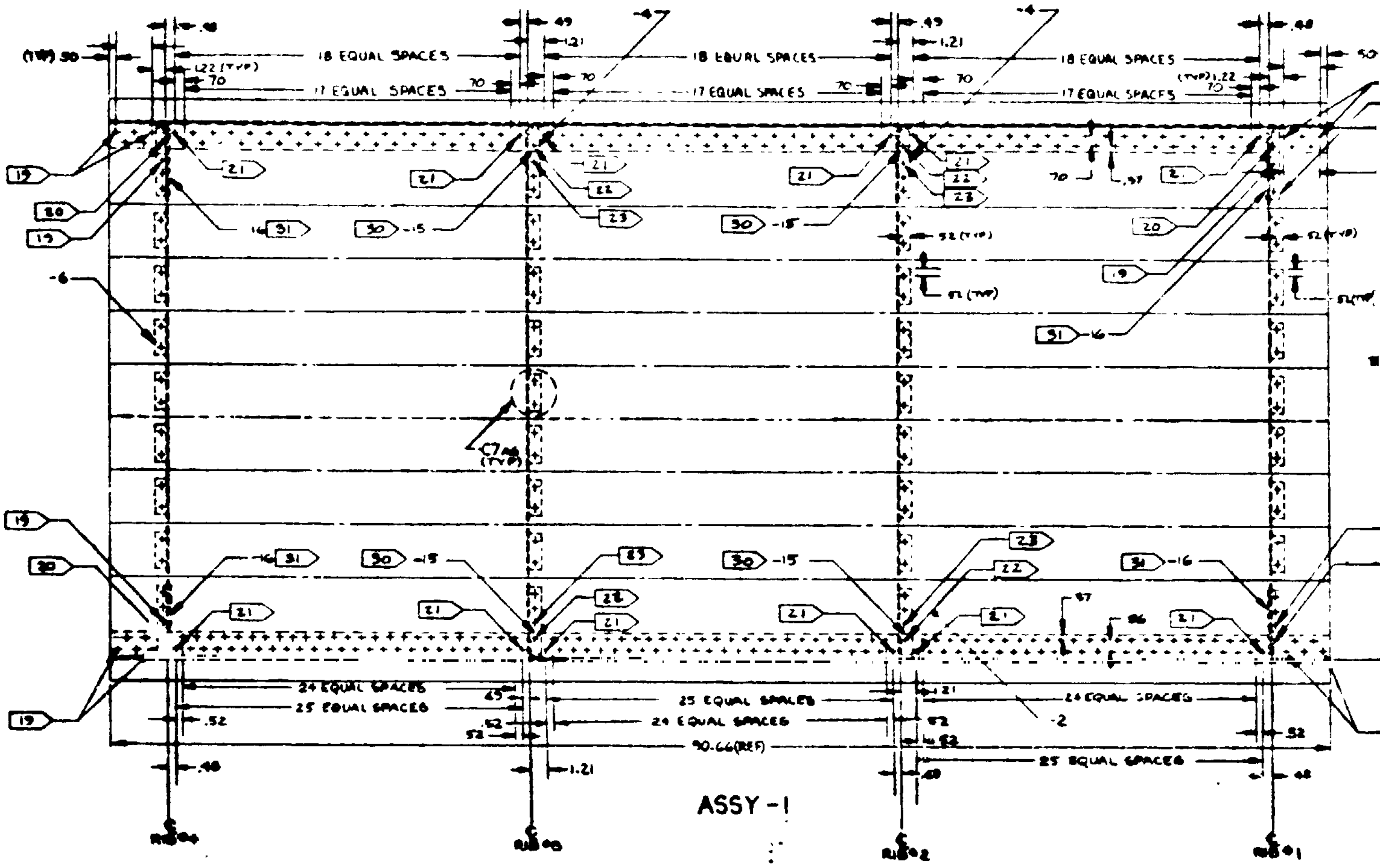
ASSY -6
(SCALE: 1/2)

65C17792

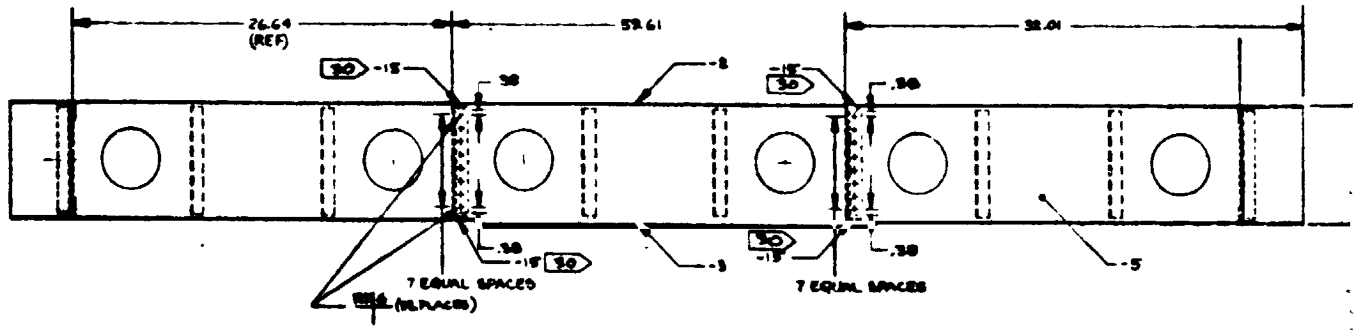
11 10 9

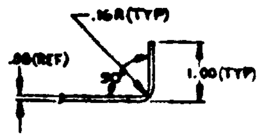
8 7 6

FOLDOUT FRAME 3



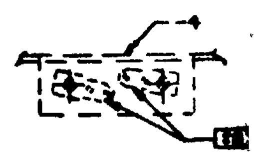
ASSY -1





B9
SCALE 1/1

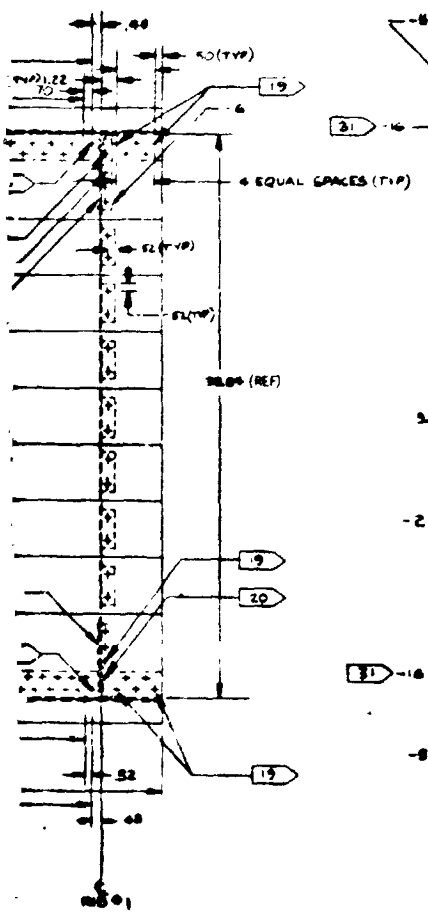
SCALE : 1/4

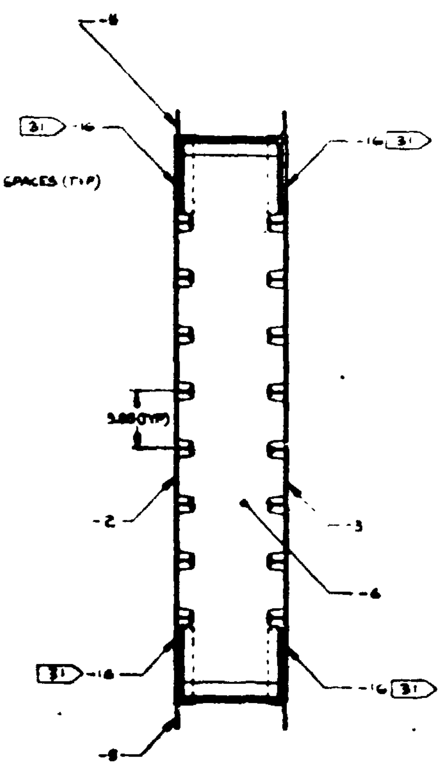


C7
(-2 PANEL ONLY)
SCALE - 1/1

8 7

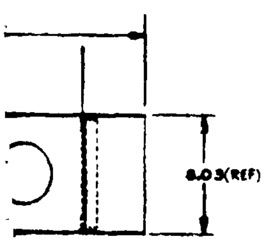
FOLDOUT FRAME





PLY TABLE

PART NUMBER	PLY NUMBER "P"	MATERIAL	TAPE OR CLOTH WARP ORIENTATION	SPLICE	REV LTR
-14	1,5,12	1	+45° OR -45°	14	
	3,8,10	1	+45° OR -45°	10	
	2,6,7	1	0°	9	
	4,9,11	1	0° OR 90°	10	
-2 PANEL 32	1,3,4,5,6,8,9,10,13,14	1	+45° OR -45°	4	
	15,16,19,20,22 & 23				
	24 3032			9	
	35,36,37,40 & 41				
	7,11,18, 17,21	1	0°	4	
	28, 34 38 & 39			9	
	2, 12,25,29	3	90°	9	
-2 PANEL S-1 THRU S-8 & -3 PANEL 17 S-1 THRU S-8	1,3,4,6 & 13	1	+45° OR -45°	4	
	2,6,5	1	0°	9	
	7,8,9,10,11 & 12	2	0°	9	
-3 PANEL 18	1,4,5, 8,9,10,11,13,14	1	+45° OR -45°	4	
	15,20,21,22 26&28				
	29,30,31,32 53			9	
	42,44,45,46, 47				
	48,49,53,54,55,56,58				
	2, 18,34,40	3	90°	9	
	3 ,17, 35,36	2	30°	9	
	6,7,12 16,19 23 24 25	1	0°	4	
	37,38,39,41,43,50,	1	0°	9	
	51,52, 57	1	0°	9	



C7
(-PANEL ONLY)
SCALE - V

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✓	40	AN960C6L	WASHER, C
✓	684	AN960C10L	WASHER,
✓	80	BACN10KB3CFM	NUTPLATE
✓	28	BACN10JC3CM	NUT
✓	40	BACC30M5	COLLAR,
✓	666	BACC30M6	COLLAR,
✓	40	BACB30NY5R4	BOLT-TI,
✓	40	BACB30NU5R3	BOLT-TI,
✓	8	BACB30NW6R5	BOLT-TI., H
✓	528	BACB30NW6R3	BOLT-TI., H
✓	8	BACB30MY6R5	BOLT-TI,
✓	140	BACB30MY6C3	BOLT-TI,
✓	8	-16	SHIM - LAM
✓	8	-15	SHIM - LAM
-4	1	-14	CORE - H
-4	2	-13	ANGLE
-5	6	-12	STIFFENER-T
-5	2	-11	ANGLE
-5	1	-10	WEB
-5	2	-9	CHORD - SPAR
-6	4	-8	ANGLE
-6	1	-7	RIB
✓	2	-6	RIB ASSEMBLY
✓	2	-5	SPAR ASS
✓	2	-4	RIB ASSEM
✓	1	-3	PANEL ASSE
✓	1	-2	PANEL ASS
—	—	-1	BOX ASSEM
-1	QTY REQD	PART OR IDENTIFYING NUMBER	NOMENCLATURE OR DESCRIPTION

65C17792

5 FOLDOUT FRAME

PLY

ALL MACHINED SURFACES.

BREAK ALL SHARP CORNERS AND EDGES ON ... DETAILS TO RADIUS OR CHAMFER OF APPROXIMATELY .010. FINISHED PARTS SHALL ... SMOOTH CUT EDGES.

PLACE AN ... WASHER UNDER ... C80M 66 COLLAR.

PLACE AN ... WASHER UNDER ... C80M 66 COLLAR.

INSTALL 5/32 DIA FASTENERS IN .1630 $\frac{.1645}{.1630}$ DIA HOLES.

INSTALL 3/16 DIA FASTENERS IN .1890 $\frac{.1925}{.1895}$ DIA HOLES.

SEE ZONE C13 FOR FLAG NOTES

ORIGINAL PAGE IS OF POOR QUALITY

NOTICE

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NOMENCLATURE OR DESCRIPTION	ZONE	MATERIAL AND SPECIFICATION	[illegible]	[illegible]	[illegible]	REV LTR
WASHER, CRES						
WASHER, CRES						
NUTPLATE						
NUT						
COLLAR, ALUM						
COLLAR, ALUM						
...T-TI, HEX-DRIVE						
...T-TI, HI-TORQUE						
...T-TI, HEX-DRIVE						
...T-TI, HEX-DRIVE						
...T-TI, HEX-DRIVE						
...LT-TI, HEX-DRIVE						
...IM-LAMINATED	...-1	BACS40R100B0F				
...IM-LAMINATED	...-1 ...-2	BACS40R10C14F				
CORE-HONEYCOMB		16				
...LE	B7-1	AND10134-1205 EXTRUSION 29		F-20 02	M	
STIFFENER-TEE		AND10136-1401 EXTRUSION 29				
		AND10133-1002 EXTRUSION 29				
		... .080 ALUM SHEET 2024-T3 PER QQ-A 250/4				
- SPAR		BAC1505-10158 EXTRUSION 29				
	A9-1	AND10133-1002 EXTRUSION 29				
	B8-1	... .081 ALUM SHEET 2024-0	T3	F-20 02		
ASSEMBLY	C6-1					
... ASSEMBLY	C5-1					
ASSEMBLY	C6-1	1				
...L ASSEMBLY	A7-1	1 2 3				
...EL ASSEMBLY	B7-1	1 2 3				
... ASSEMBLY	B7-1				M	

LIST OF MATERIAL

A-8

CONTRACT NO. EWA2211KT

ADVANCED COMPOSITES

DRAWN F. CHALKLEY

CHECKED C. ...

THE BOEING COMPANY

COMMERCIAL AIRPLANE GROUP, SEATTLE, WASH

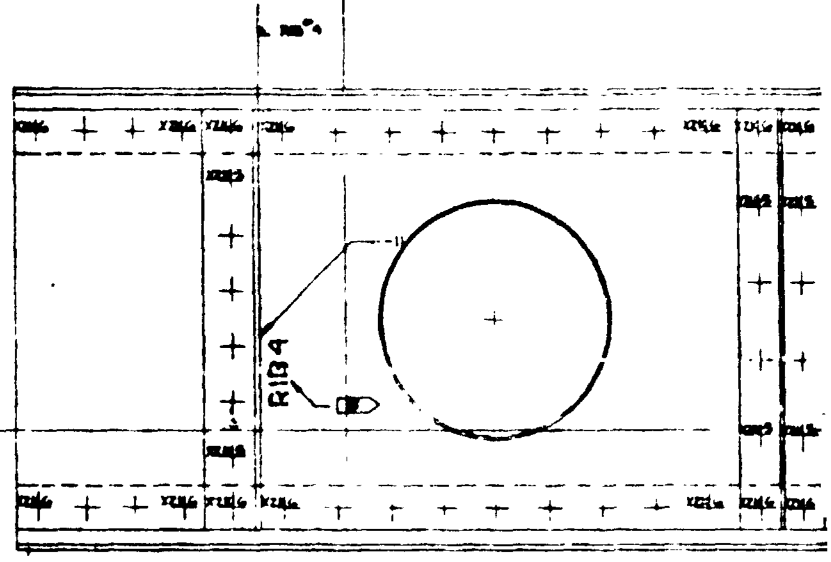
NASA BOX ASSY
SONIC TEST #20
GR/EP STABILIZER

65C17792

FOLDOUT FRAME

ORIGINAL PAGE IS
OF POOR QUALITY

D



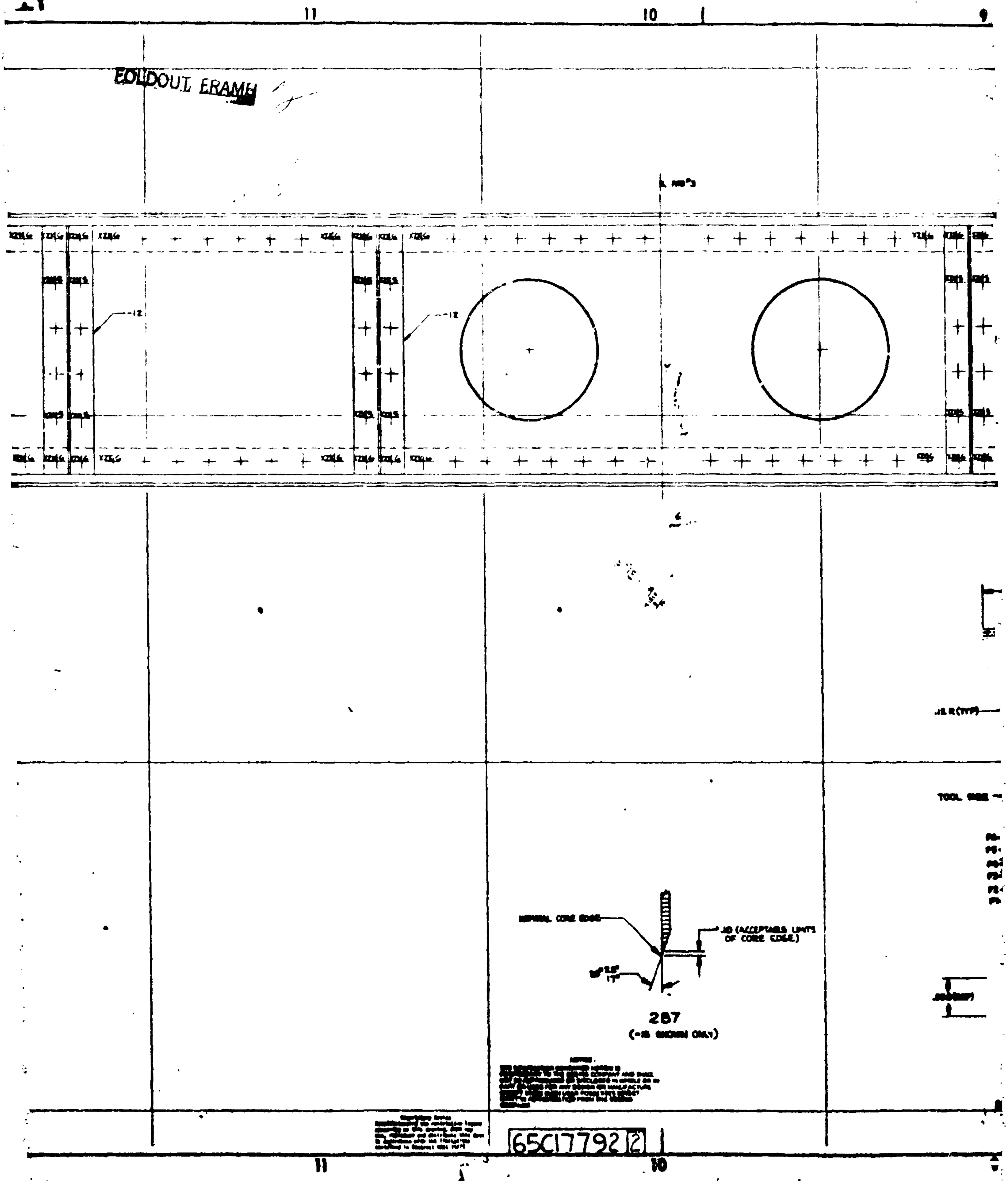
C

B

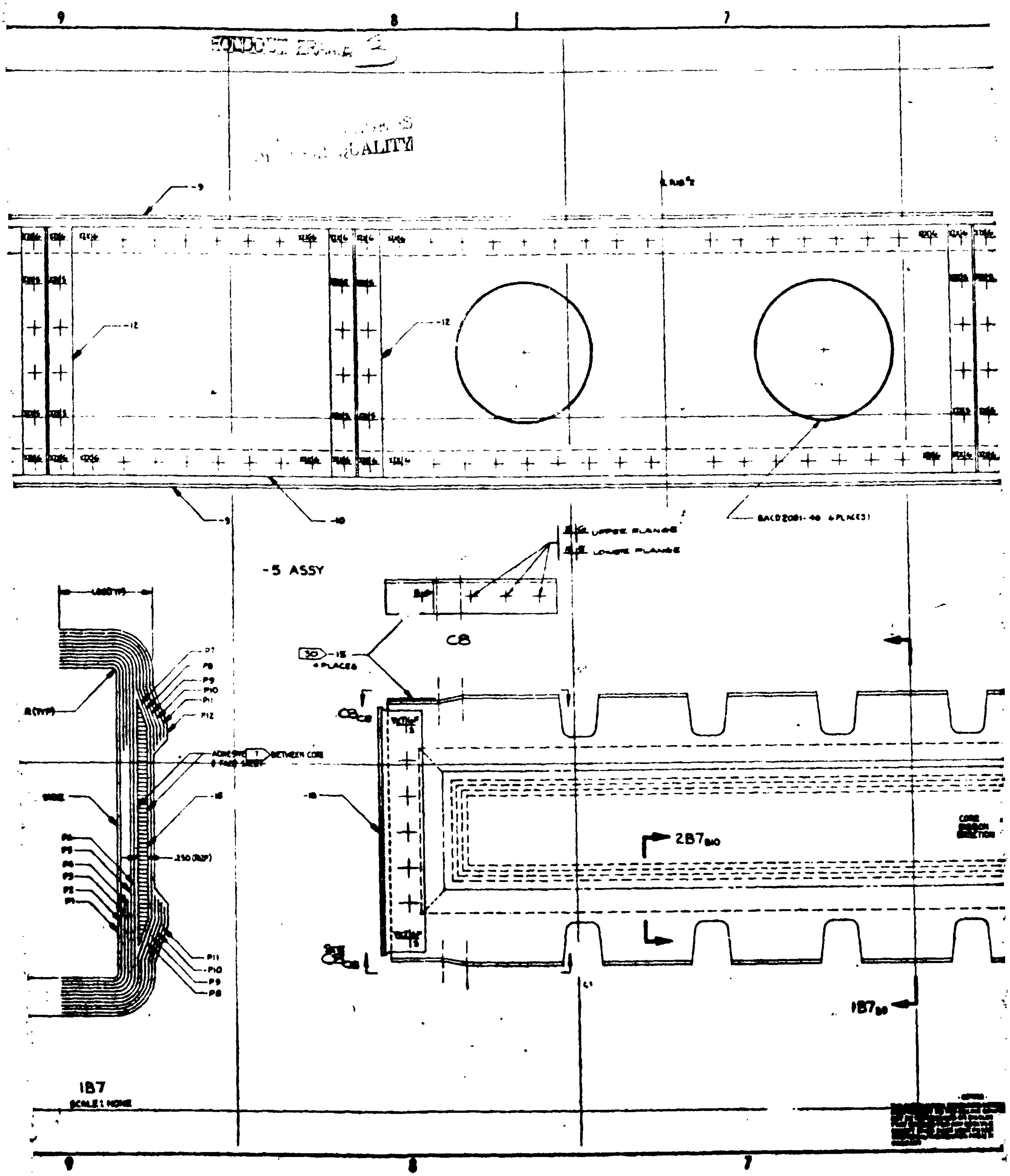
13

12

FOLDOUT FRAME

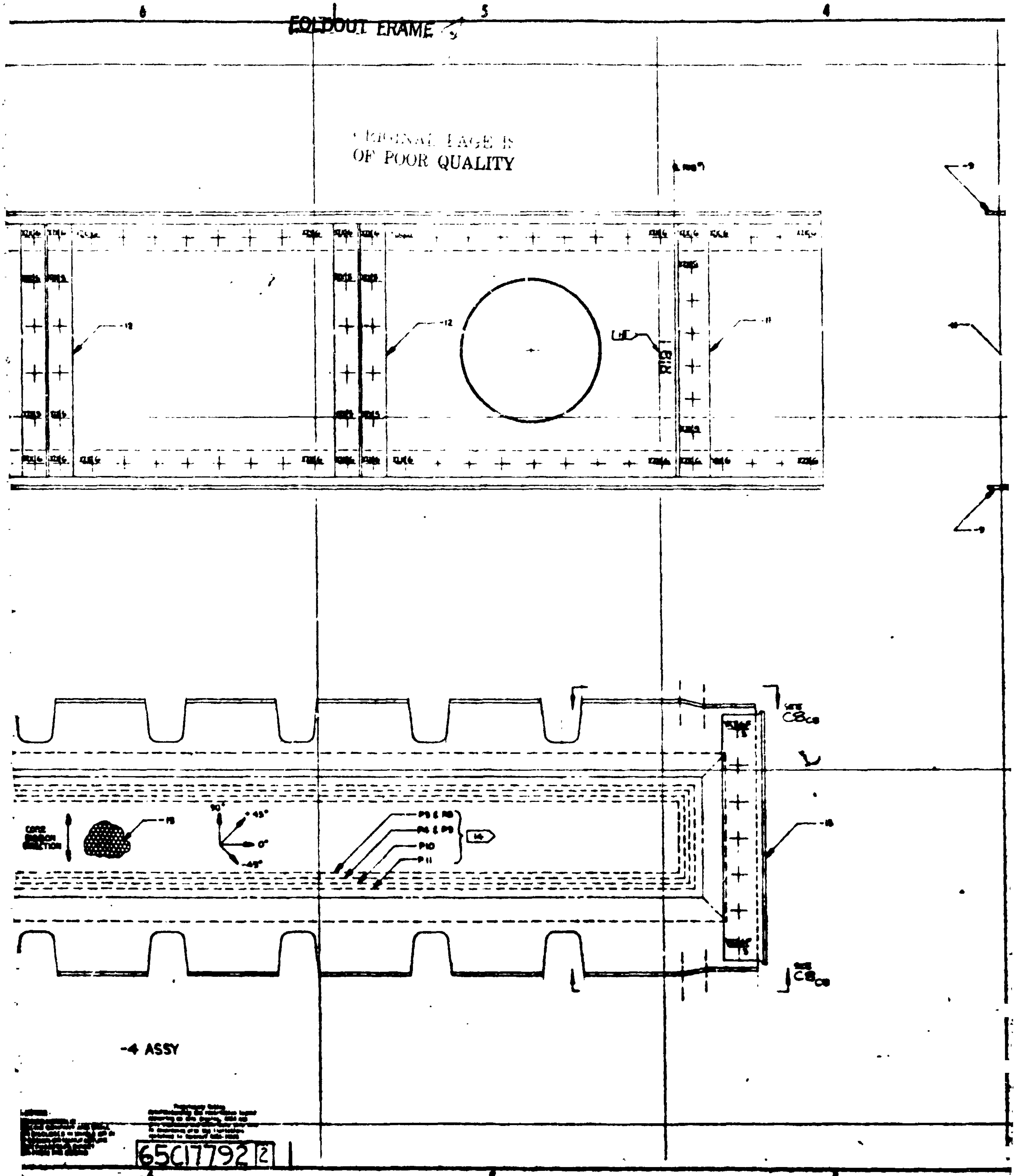


65C17792 2



9
8
7
FOLDOUT FRAME 2
-5 ASSY
BACD2081-40 6 PLACES
UPPER FLANGE
LOWER FLANGE
C8
30
-15
4 PLACES
2B7
1B7
CORE RIBBON DIRECTION
ADHESIVE 1 BETWEEN CORE & FACE SHEET
.250 (REF)
R (TYP)
P7
P8
P9
P10
P11
P12
P4
P5
P6
P3
P2
P1
-16
-18
-12
-9
-10
1B7
SCALE: NONE

ORIGINAL PAGE IS
OF POOR QUALITY



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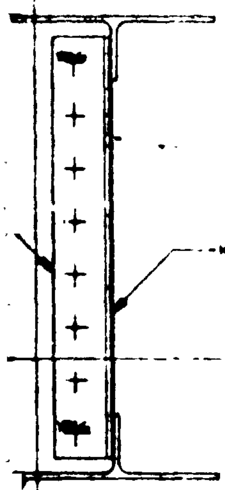

A-9

NASA BOX ASSY-
SONIC TEST-2C
GR/EP STABILIZER

65C17792

65C17792

2

PCM

FOLDOUT FRAME

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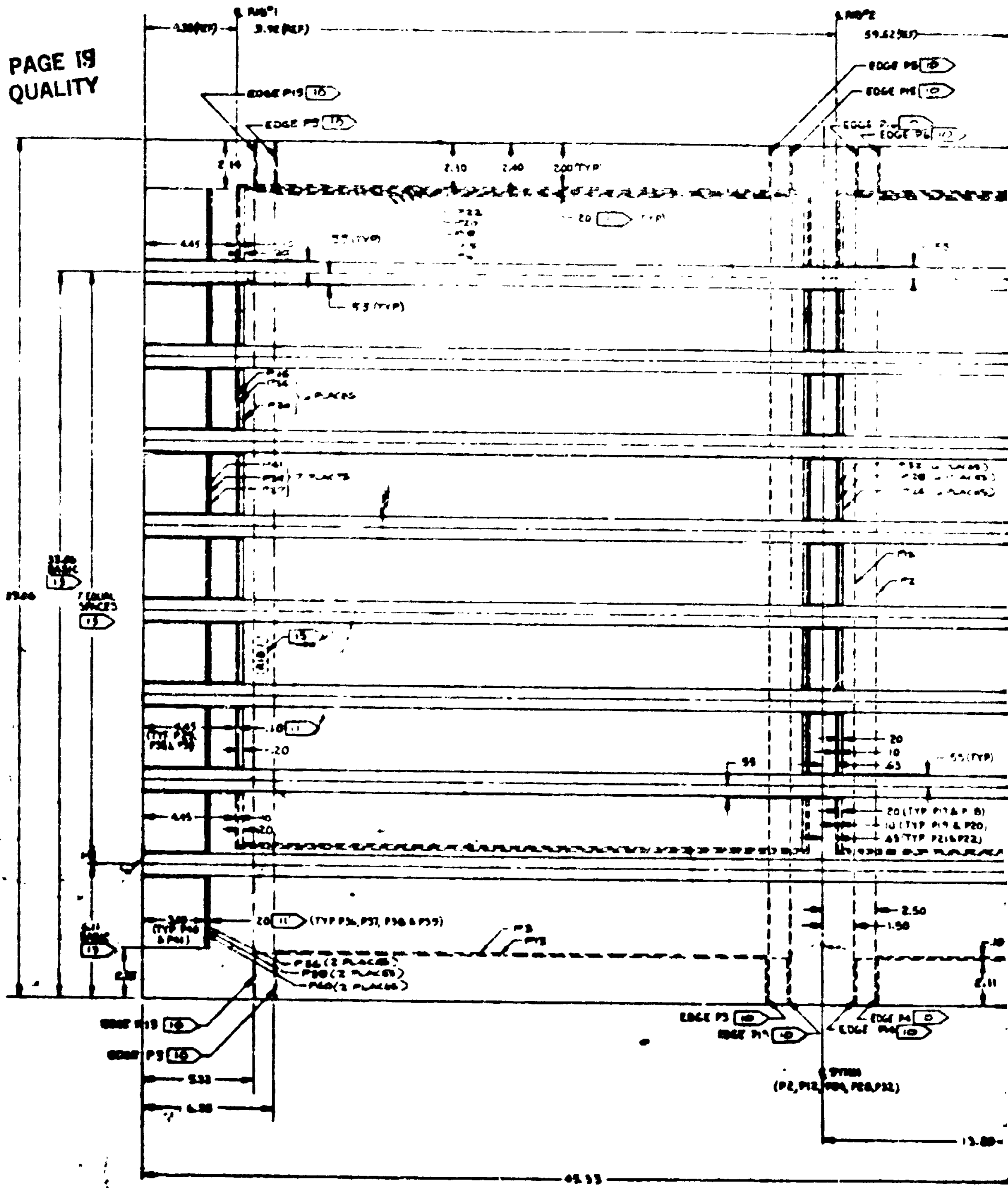
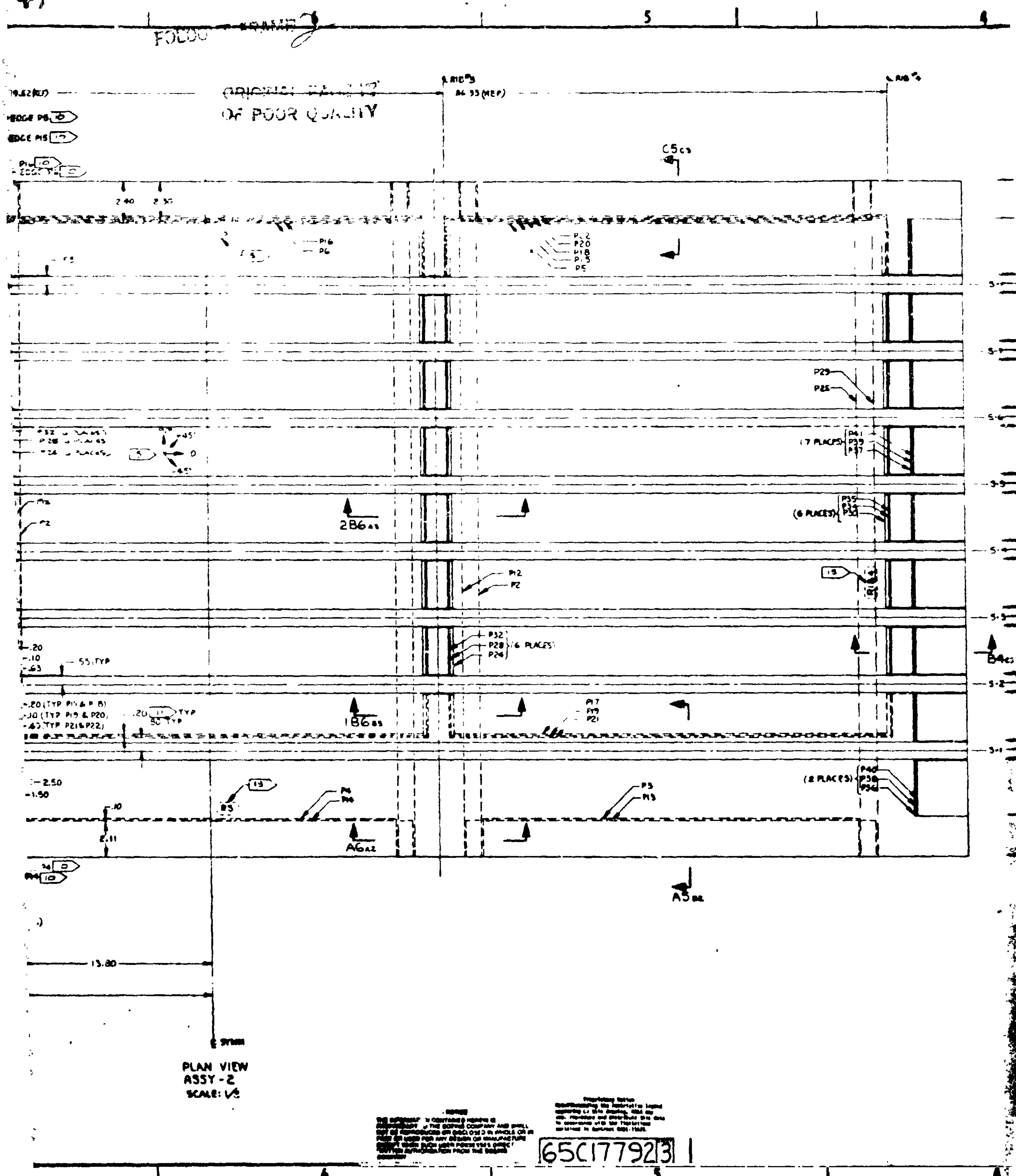
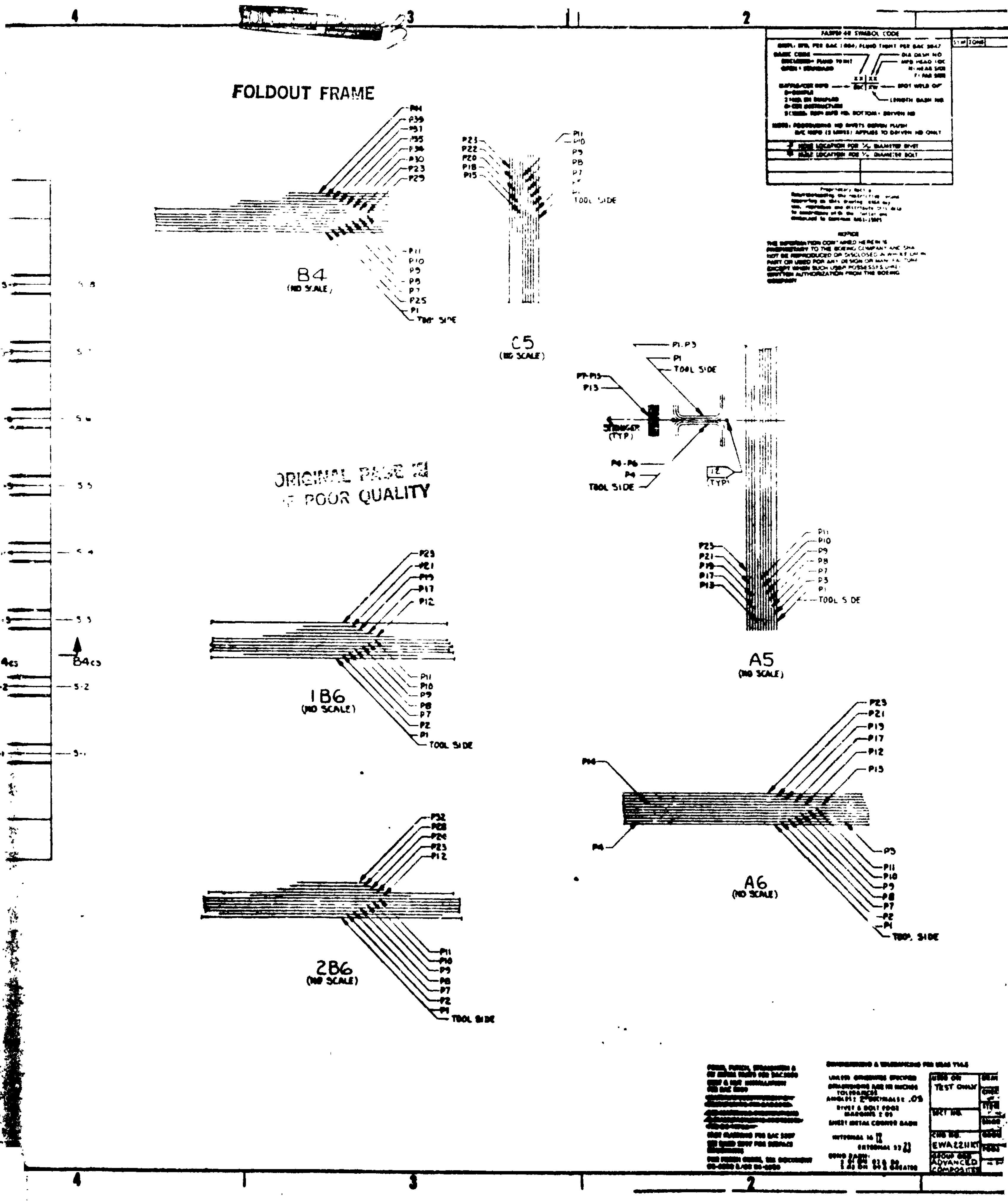



ORIGINAL PAGE IS
OF POOR QUALITY
℄ RIB 3
℄ RIB 4
C5
B4
A5
2B6
1B6
A6
(7 PLACES)
(6 PLACES)
(2 PLACES)
P32
P28
P24
P12
P2
P17
P19
P21
P5
P15
P16
P6
P29
P25
P41
P39
P37
P35
P34
P30
P40
P38
P36
13.80
.55 TYP
2.50
1.50
2.11
.10
2.40
2.30
℄ SYMM
PLAN VIEW
ASSY -2
SCALE: 1/2
65C17792 3

FOLDOUT FRAME

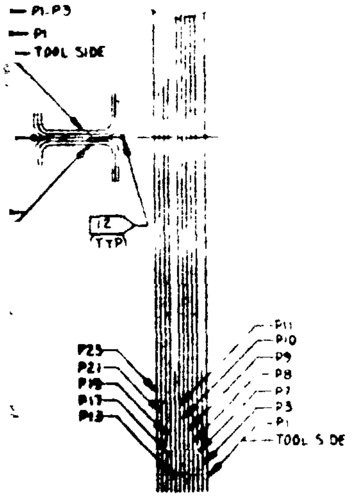


FOLDOUT FRAME 4

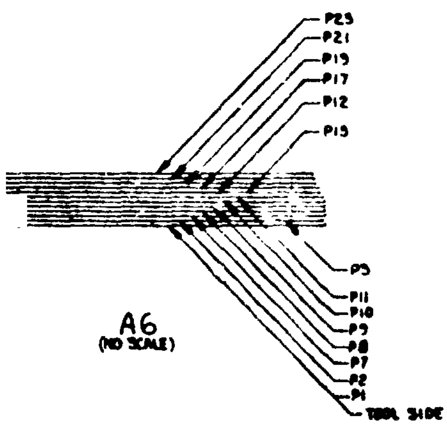
FASTENER SYMBOL CODE

REVISIONS

SYM	ZONE	DESCRIPTION	DATE	APPROVAL



A5
(NO SCALE)



A6
(NO SCALE)

A-10

65C17792

SH 3

TEST ONLY	CHAMLEY		THE BOEING COMPANY
			NASA BOX ASSY- SONIC TEST #20, GR/EP STABILIZER
EWA22HKT			J 65C17792
ADVANCED COMPOSITES			SCALE 1/2

65C17792 SH 3

FOLDOUT FRAME

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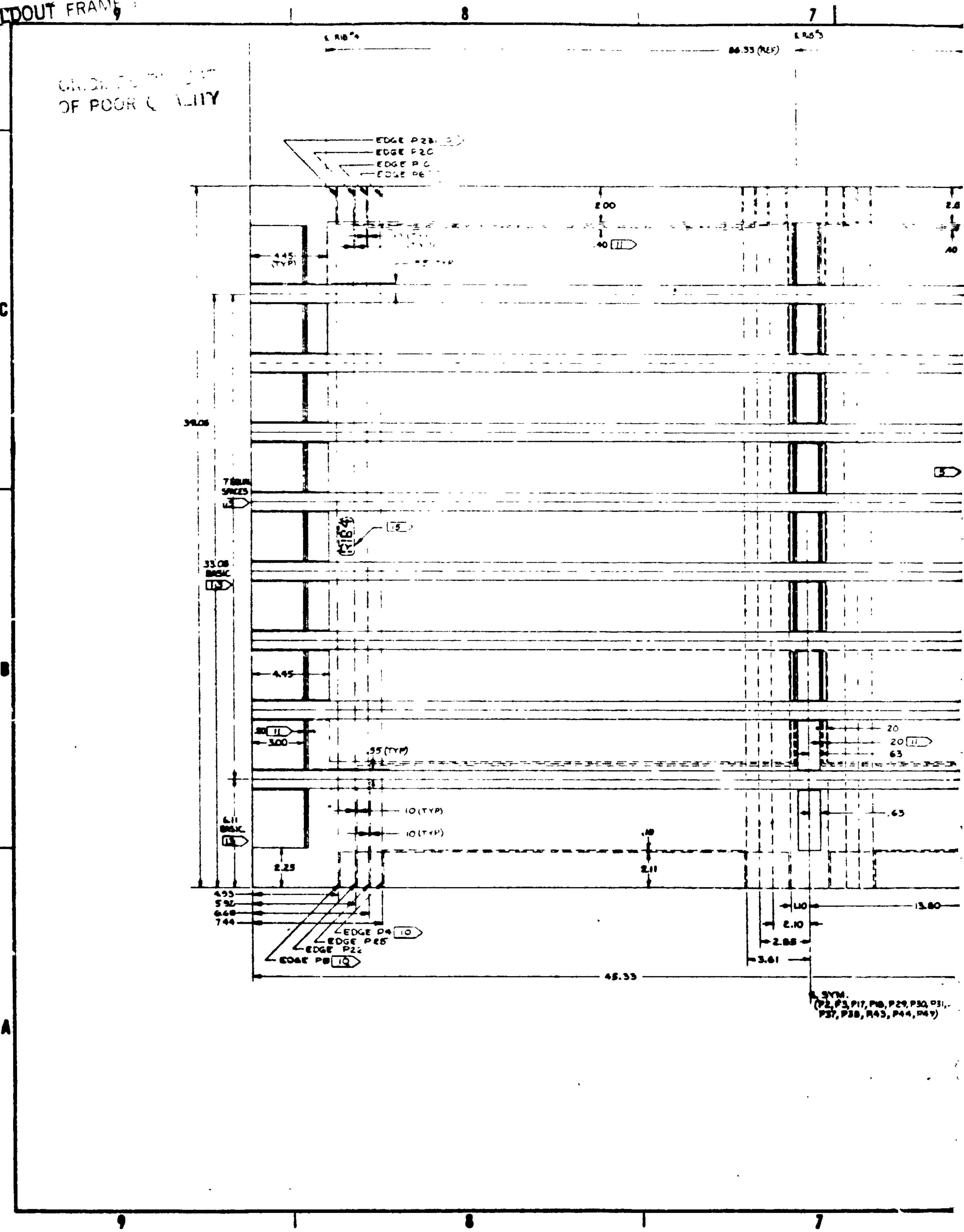

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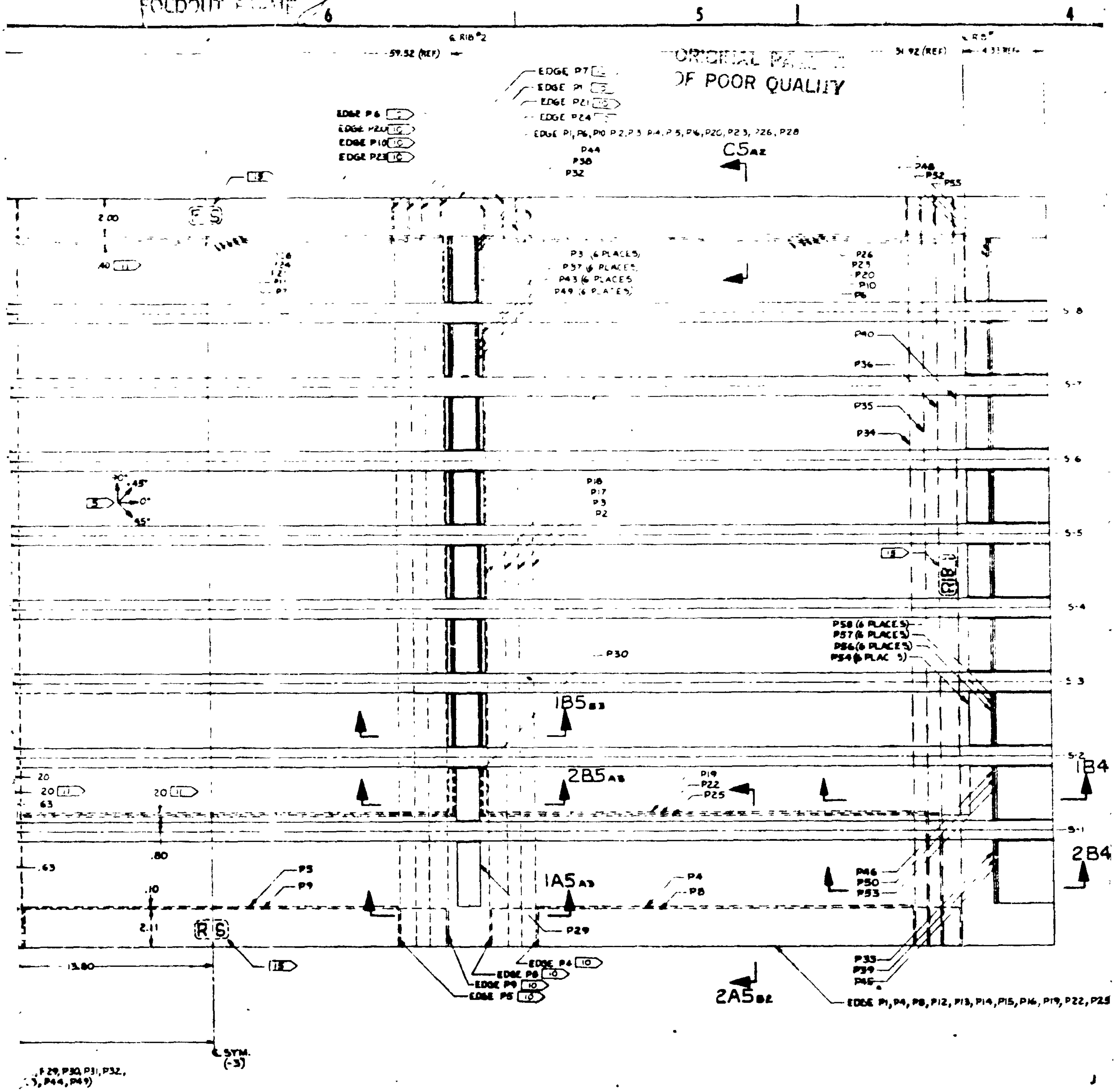

PLAN VIEW
ASSY -3
SCALE 1/2

65C17792 4

FOLDOUT FRAME

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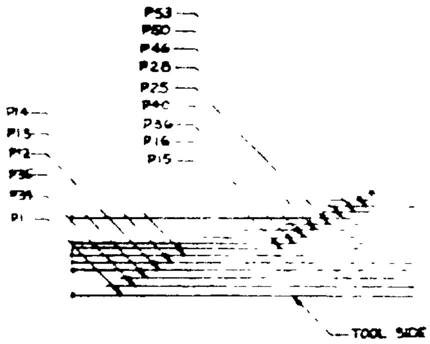
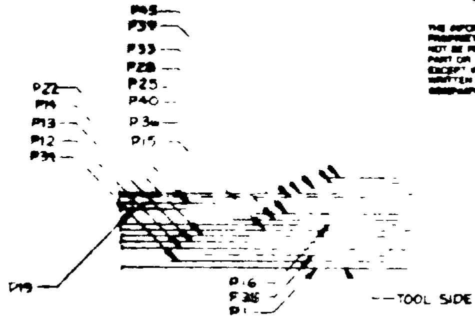
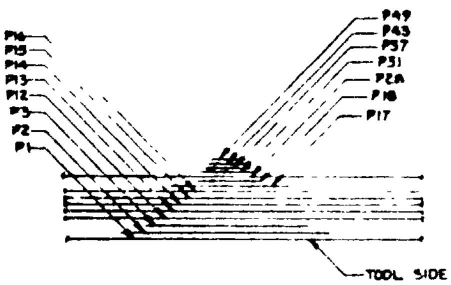


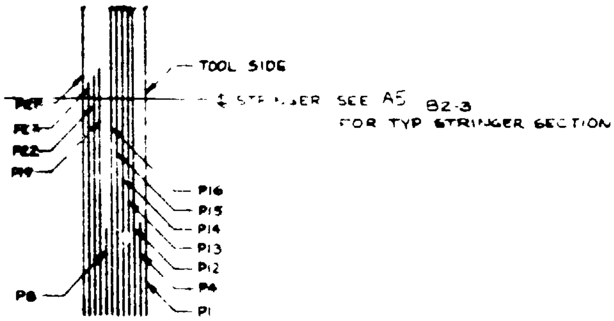
1B4
(NO SCALE)



2B4
(NO SCALE)



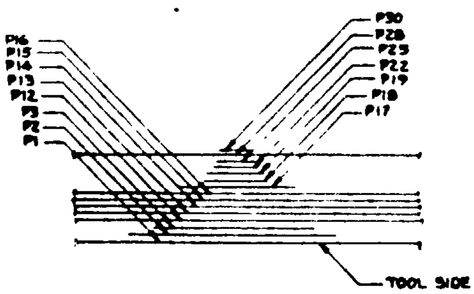
1B5
(NO SCALE)



2A5
(NO SCALE)

1B4
2B4

, P19, P22, P25, P28



2B5
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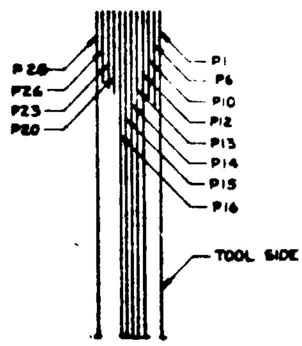

C5
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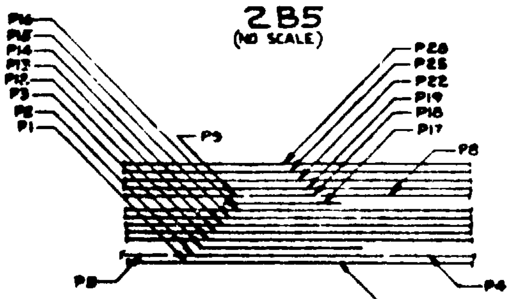

1A5
(NO SCALE)

TEST ONLY

FOLDOUT FRAME

FASTENER SYMBOL CODE

HOLE LOCATION FOR DIAMETER RIVET
HOLE LOCATION FOR DIAMETER BOLT

NOTICE

THE INFORMATION CONTAINED HEREIN IS PROPRIETARY TO THE BOEING COMPANY AND SHALL NOT BE REPRODUCED OR DISCLOSED IN WHOLE OR IN PART OR USED FOR ANY DESIGN OR MANUFACTURE EXCEPT WHEN SUCH USER POSSESSES DIRECT WRITTEN AUTHORIZATION FROM THE BOEING COMPANY

TOOL SIDE

ORIGINAL PAGE IS OF POOR QUALITY

C

SEE A5 B2-3 FOR TYP STRINGER SECTION

B

4

65C17792

A

A-11

USED ON	TEST ONLY
DRAWN	C.R. HELGESON
	THE BOEING COMPANY COMMERCIAL AIRPLANE DIVISION RENTON, WASH
TITLE	NASA BOX ASSY — SONIC TEST # 20, GR/EP STABILIZER
CODE IDENT NO	81205
SIZE	J
NO.	65C17792
SH	4

65C17792 SH 4

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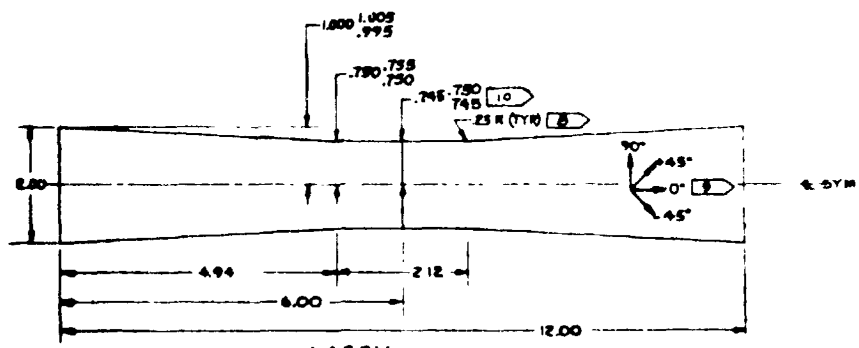

-1 ASSY
-2 ASSY
-3 ASSY

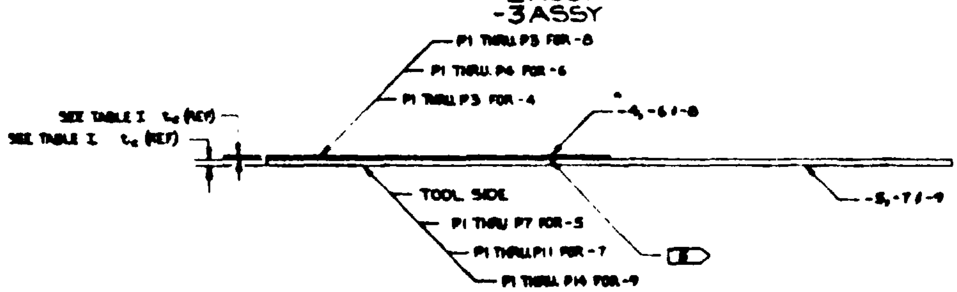

TABLE I		
ASSY No.	DETAIL PART No.	t_c (REF)
-1	-4	.0225
-1	-5	.0525
-2	-6	.0300
-2	-7	.0785
-3	-8	.0225
-3	-9	.1050

FOLDOUT FRAME

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ALL MACHINED SURFACES EXCEPT AS NOTED.

PLY TABLE

PART No.	PLY No "P"	MAT'L & SPEC	CLOTH WARP ORIENTATION	SPLICE	REV TR
-4 & -8	1 & 3	1	+ OR - 45°	7	
	2	1	0°	6	
-5	1,3,4,5 & 7	1	+ OR - 45°	7	
	2 & 6	1	0°	6	
-6	1 & 4	1	+ OR - 45°	7	
	2 & 3	1	0°	6	
-7	1,5,6,7 & 11	1	+ OR - 45°	7	
	2 & 10	2	0°	6	
	3 & 9	3	0°	6	
	4 & 8	1	0°	6	
-9	1,3,4,5,6, 7,8,10,12 & 14	1	+ OR - 45°	7	
	2,6,9,11 & 13	1	0°	6	

1 EPOXY PREIMPREGNATED GRAPHITE WOVEN FABRIC PER BMS8-212 TYPE II, CLASS 2, STYLE 3K-70-P. FABRICATE PER BAC5562.

2 EPOXY PREIMPREGNATED GRAPHITE UNIDIRECTIONAL TAPE BMS8-212, TYPE II, CLASS 1, GRADE 190. FABRICATE PER BAC5562.

3 EPOXY PREIMPREGNATED GRAPHITE UNIDIRECTIONAL TAPE BMS8-212, TYPE II, CLASS 1, GRADE 145. FABRICATE PER BAC5562.

4

5 COCURE -4 TO -5, -6 TO -7, -8 TO -9 PER BAC5562.

6 NO SPLICES ALLOWED

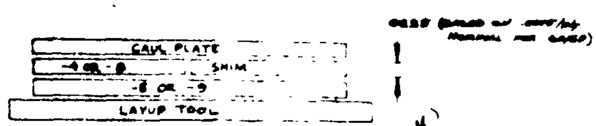
7 LAP SPLICE PER BAC5562

8 ALL CONNECTING CURVED SURFACES OR CURVED AND PLANE SURFACES SHOWN AS TANGENT MUST BE BLENDED SMOOTHLY

9 PLY ORIENTATION CONVENTION: FABRIC 0° IS PARALLEL TO THE WARP DIRECTION. TAPE 0° IS PARALLEL TO THE FIBER DIRECTION

10 THE WIDTH OF THE TEST SECTION OUTWARD FROM .745 SHALL INCREASE GRADUALLY AND EQUALLY ON EACH SIDE UP TO .750 SO THAT NO ABRUPT CHANGES IN DIMENSION OCCUR.

SUGGESTED METHOD OF COCURE FOR -4 & -5 ASSYS:



℄ SYM

				BAC5562	PROCESS SPEC.						
✓			1	-9	GRAPHITE-EPOXY PLATE		1	—			
✓			1	-8	↑		1	—			
	✓		1	-7	↕		1, 2 & 3	—			
	✓		1	-6	↕		1	—			
		✓	1	-5	↓		1	—			
		✓	1	-4	GRAPHITE-EPOXY PLATE		1	—			
—			—	-3	TEST SPECIMEN ASSY.			—		M	
	—		—	-2	TEST SPECIMEN ASSY.			—		M	
		—	—	-1	TEST SPECIMEN ASSY.			—		M	
-3	-2	-1	QTY REQD	PART OR IDENTIFYING NUMBER	NOMENCLATURE OR DESCRIPTION	ZONE	MATERIAL AND SPECIFICATION	WT TO	FINISH	PT-INS	REV LTR

EWA 2211MT

ADV. COMP B-747C

65C17980

FOLDOUT FRAME

FOLD

ORIGI...
OF PO...

EXCEPT AS NOTED.

TAPE

TAPE

A-12

	TEST ONLY	—					
NEXT ASSY	USED ON		SERIAL NUMBER	PART NUMBER		DRAWING SHEET NUMBER	REV LTR
APPLICATION							

CHANGE NO
EWA 2211MT

ADV. COMP
B-7410

DRAWN L.A. HELGESON

THE BOEING COMPANY
COMMERCIAL AIRPLANE DIVISION RENTON WASH.

NASA TEST SPECIMEN
PANEL DETAIL TEST #22,
GR/EP STABILIZER

SIZE J

65C17980

65C17980

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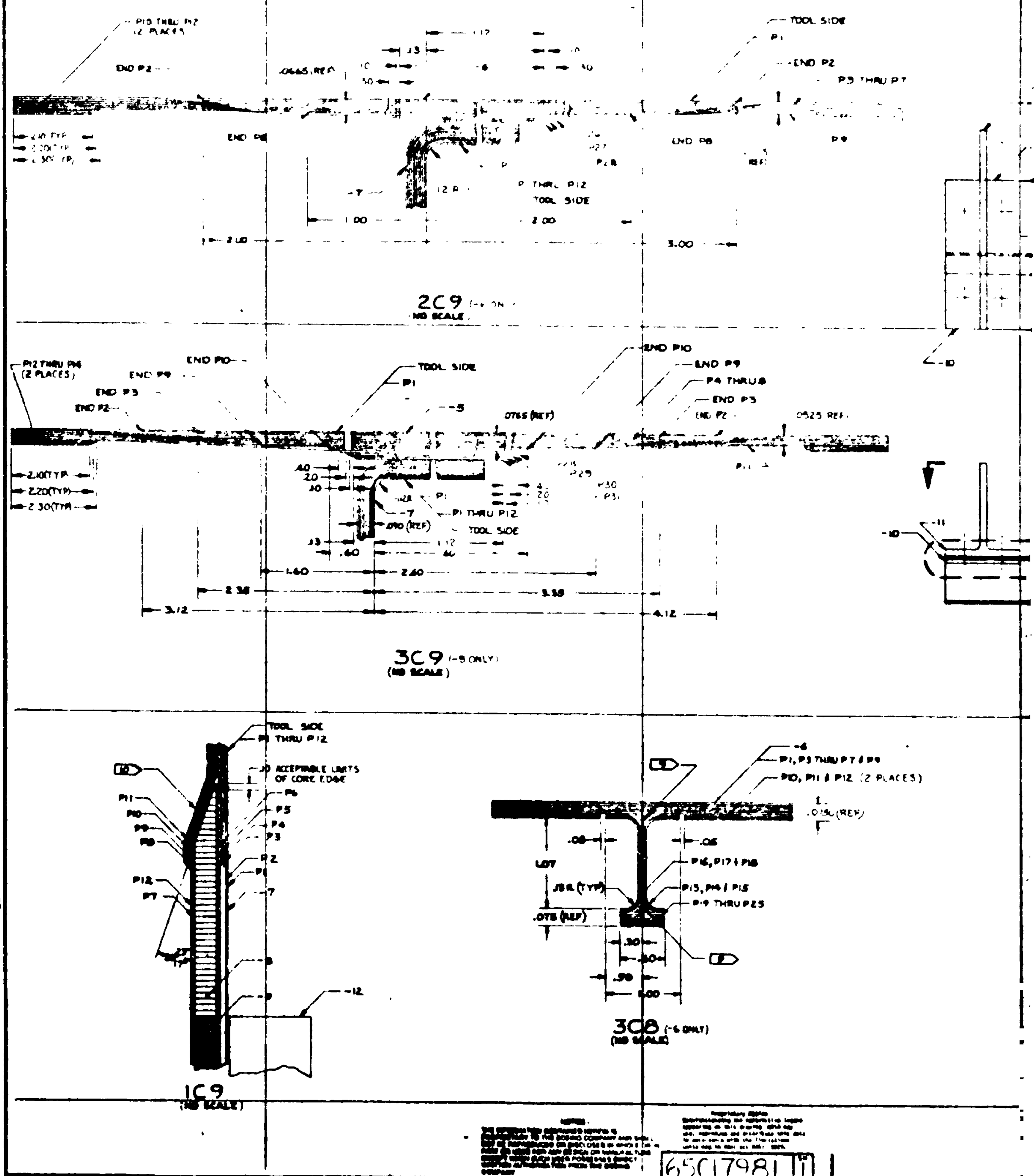

12
11
10

FOLDOUT FRAME

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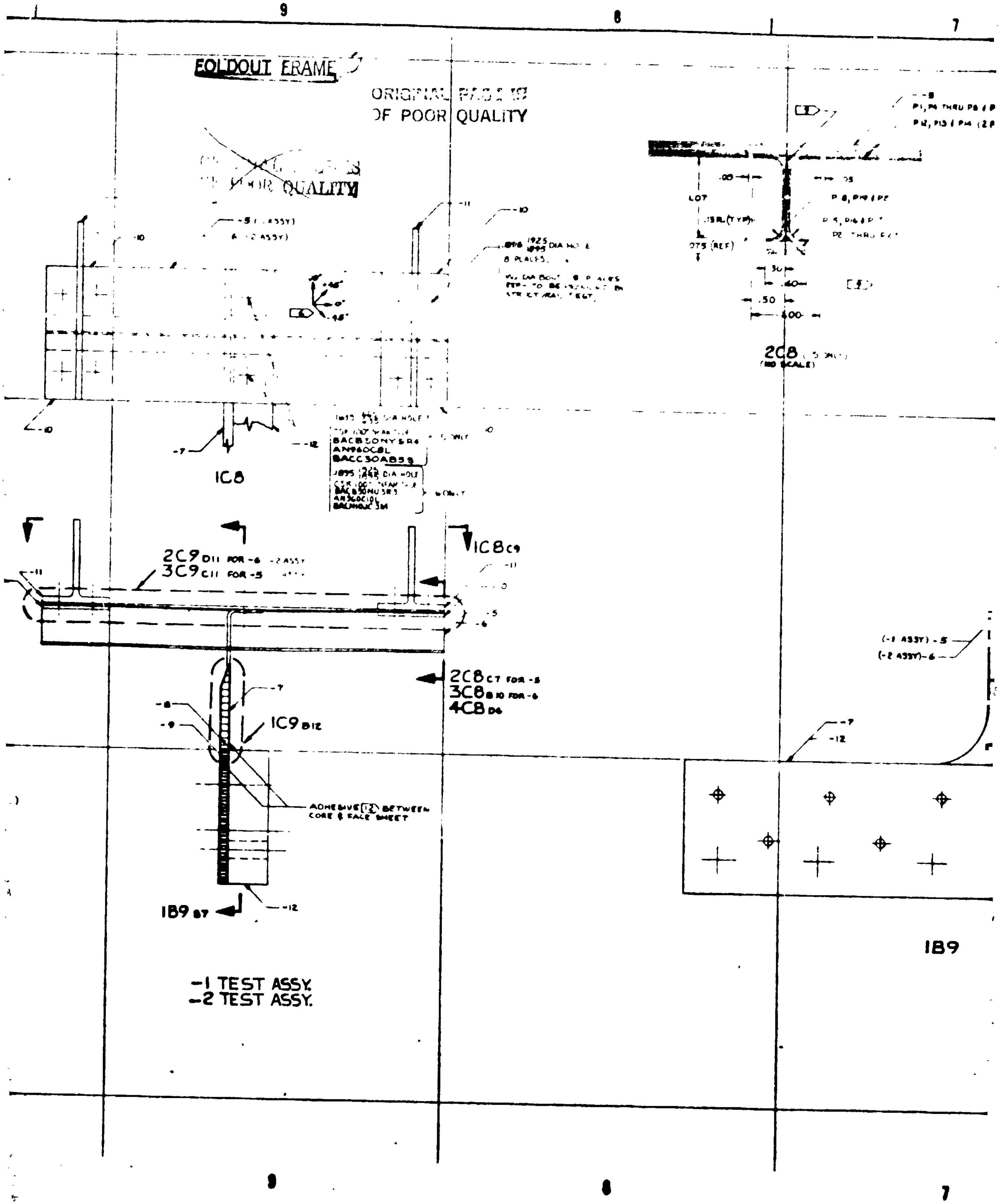

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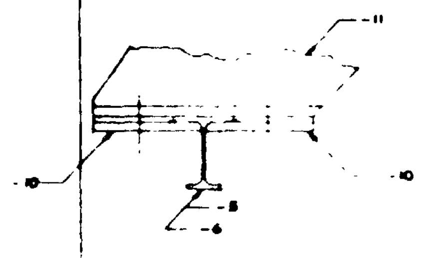
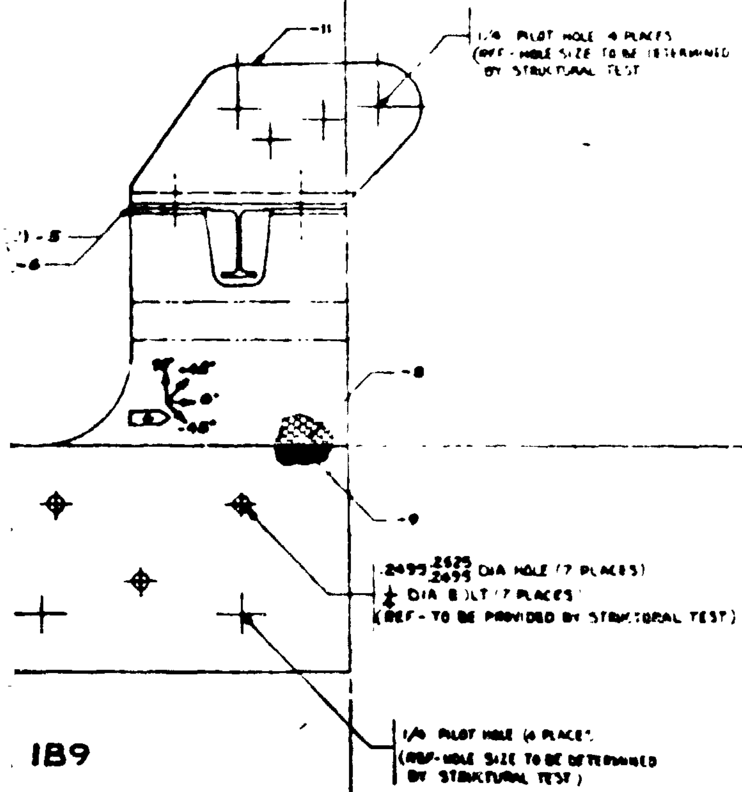


4C8

PLY TABLE

PART No.	PLY No "P"	MAT'L	... WARP ORIENTATION		
-5	1,5,6,7,11,12,14, 15,17,18,20,22,23	[illegible]	... 45°	[illegible]	
	2 & 10	[illegible]		2	
	3,9,21 THRU 26	[illegible]		4	
	4,8,13,16,19,27	[illegible]	0°	3	
-6	1,4,5,6,9,10,12 13,14,18,25,26,28	[illegible]			
	3,7,11,14,17,27	[illegible]			
	2,8	[illegible]			
	19 THRU 24	[illegible]			
-7	1,3,12	[illegible]	... 45°		
	5,8,10	[illegible]			
	2,6,7	[illegible]	0°		
	4,9,11	[illegible]			



1B9

PLY

ALL MACHINE...

BREAK ALL SHARP ... TO A RADIUS OR CHA...

1. EPOXY PREIMPREG... BMS 8-212, TY... FABRICATE PER...
2. EPOXY PREIMPREG... BMS 8-212, TYPE... FABRICATE PER...
3. EPOXY PREIMPREG... BMS 8-212, TYPE... FABRICATE PER...
4. POLYPLUGS A...
5. LAP SPLICE ...
6. PLY ORIENTATION... PARALLEL TO T... PARALLEL TO T...
7. HONEYCOMB CO... GRADE 3.0 (1/8...
8. HONEYCOMB CORE...
9. GRAPHITE FILLER... UNIDIRECTIONAL... OPTIONAL PER...
10. APPLY ONE LAY... 100 BB 30TR ...
11. 2024-T3511 ... OPTIONAL: 707...
12. COCURE WITH AM... CLASS I, GRADE... PER BAC 5562

		✓	✓	1
		✓	✓	2
		✓	✓	4
-7	-7			1
-7	-7			1
✓	✓			1
✓				1
	✓			1
—		✓		1
	—		✓	1
		—		—
			—	—
-4	-3	-2	-1	QTY REQD

65C17981

4 3 2

PLY

F ALL MACHINED SURFACES

R ALL SHARP CORNERS & EDGES ON GRAPHITE DETAILS RADIUS OR CHAMF OF APPROX .010

EPOXY PREIMPREGNATED GRAPHITE WOVEN FABRIC PER BMS8-212, TYPE II, CLASS 2 STYLE
FABRICATE PER BAC5562

EPOXY PREIMPREGNATED GRAPHITE UNIDIRECTIONAL TAPE BMS8-212, TYPE II, CLASS 1, GRADE 19
FABRICATE PER BAC5562.

EPOXY PREIMPREGNATED GRAPHITE UNIDIRECTIONAL TAPE BMS8-212, TYPE II, CLASS 1, GRADE 145
FABRICATE PER BAC5562

NO SPLICES ALLOWED

LAP SPLICE PER BAC5562

PLY ORIENTATION CONVENTION FABRIC PARALLEL TO THE WARP DIRECTION PARALLEL TO THE FIBER DIRECTION

HONEYCOMB CORE, BMS8-124, CLASS IV, TYPE V, GRADE 3.0 (1/8 IN CELL NOMEX)

HONEYCOMB CORE BMS8-124 CLASS I, TYPE I, GRADE 12

GRAPHITE FILLERS AS REQUIRED EPOXY IMPREGNATED GRAPHITE UNIDIRECTIONAL TAPE, TYPE II, CLASS I, GRADE 95, 145 OR 190 OPTIONAL PER BMS8-212, FABRICATE PER BAC5562.

APPLY ONE LAYER OF TEDLAR FILM (BONDABLE PVF) TRANSPARENT 100 BG 30TR PER BAC5562 ON THE NON-TOOL SIDE.

2024-T3511 PER QQ-A-200/3
OPTIONAL: 7075-T6511 PER QQ-A-200/11

CURE WITH AMERICAN CYANAMID FM300 BMS 8-245 TYPE II CLASS 1 GRADE 05 ADHESIVE BETWEEN CORE & FACE SHEETS PER BAC 5562

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✓	1	-12	DOUBLER		120 x 40 x 130 7075 T6 BARE PLATE QQ-A-250/12		F-2002	M	
✓	2	-11	FITTING		MAKE FROM BAC 505-10025		F-2002	M	
✓	4	-10	RADIUS FILLER		MAKE FROM BAC 1512-3369-20		F-2002	M	
	1	-9	CORE				—	—	
	1	-8	CORE				—	—	
	1	-7	RIB				F-2501	M	
	1	-6	PANEL					M	
	1	-5	PANEL					M	
	1	-4	TEST SPECIMEN					M	
	1	-3	TEST SPECIMEN					M	
	—	-2	TEST ASSY					M	
—	—	-1	TEST ASSY				F-2501	M	
	QTY REQD	PART OR IDENTIFYING NUMBER	NOMENCLATURE OR DESCRIPTION	ZONE	MATERIAL AND SPECIFICATION				

—	TEST ONLY	—	
		APPLICATION	

NASA TEST PANEL TO RI GR EP STA

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OF POOR QUALITY

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2

D

C

B

65C17981

A-13

—	TEST ONLY	—					
NEXT ASSY	USED ON	SECT NO	SERIAL NUMBER	PART NUMBER		DRAWING SHEET NUMBER	REV LTR
APPLICATION							

CHANGE NO
EMA 2211NT

COMR
87410

THE BOEING COMPANY
COMMERCIAL AIRPLANE DIVISION RENTON WASH.

NASA TEST SPECIMEN - SKIN PANEL TO RIB JOINT TEST # 24 GR/EP STABILIZER

J 65C17981

2

PCM